Cranfield UNIVERSITY

DIMITRIOS TOUFEXIS

MSc by Research

Aircraft Maintenance and Development of a Performance-based Creep Life Estimation for Aero Engine

Academic Year: 2010 - 2011

SCHOOL OF ENGINEERING
Department of Aerospace Sciences

Supervisor: Dr. AL SAVVARIS

January 2012

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Executive summary

For any machine designed to generate power, or to fulfill its functions in general, maintenance actions will have an impact on many aspects of its overall capabilities, especially its performance and the length of its useful life. Since these are vital in order to generate maximum profit, the maintenance actions that affect them must be given serious consideration. For this reason, this research aims to propose a method that will enhance the cost saving potential with more accurately determined maintenance intervals and greater exploitation of the remaining life of the components by utilizing the capabilities of condition based monitoring.

Initially, the research focuses on the description and the understanding of maintenance methods as they are performed within the aviation industry, but it also aims to investigate the state of the art Condition Based Monitoring Maintenance (CBMM) and its associated advantaged relating to the older methods. The thesis begins by describing the fundamental aviation maintenance management domains, paying particular attention to CBMM, and continues with the diagnostic and prognostic methods that are in use in order to support the condition monitoring concept. Next, a description is given of the actual implementations of the CBMM process, with the presentation of the maintenance enhancement systems, namely the Central Maintenance System and the Aircraft Condition Monitoring System.

Lastly, a case study is presented of the estimation of the remaining useful life of a turbine blade, as it relates to the primary failure mode of creep. The case study endorses the use of the condition monitoring diagnostic methods discussed previously and also aims to demonstrate the predictive capabilities of the Engine Usage Diagnostics at both the design and the into-service stage. The created/simulated engine performance models concern several operating conditions of the engine while the impact of each of those on the remaining useful life of the blade is investigated.

The benefit of this research is that it proposes a practical, effective, and relatively easy way to perform maintenance by predicting the need according to the usage. Additionally, the data required have already been measured, which paves the way for the creation of more intelligent engine control units. The contribution and innovation of the research is demonstrated by the fact that no similar approaches to creep life prediction have been published for the same type of engine, namely the CFM56 5B2. Last but not least, the results are presented in the most beneficial form of remaining hours before the failure.

Aims and Objectives

The aim of this project is to investigate Condition Based Monitoring (CBM) techniques and the implementation of them in aviation maintenance. The resulting maintenance by using these techniques is known as Condition Based Monitoring Maintenance (CBMM). In order to demonstrate the concept of CBMM, a case study utilizing creep life analysis for high pressure turbine blades will be considered to predict and estimate the remaining useful life of the component. Additionally, and because this particular study concerns the design lifing approach, a wide range operation scenarios will be presented in order to arrive at suggestions for health management operation.

The final aim will be achieved by completing the following objectives:

I. Review of maintenance practices and methods with particular emphasis on condition based monitoring.

II. Analysis of diagnostic/prognostic methods for aircraft engines that are in use as tools for assessments regarding the condition of the engine.

III. Description and analysis of on-board/off-board maintenance enhancement systems.

IV. Case study: Calculation of remaining useful life of a turbine's blade using design lifing approach and investigation of the impact of different ambient conditions and reduced take-off thrust health management operation.

V. Discussion and conclusions

Acknowledgements

This section is dedicated to those whose contribution was significant in the fulfillment of this MSc. The first person I must mention, and for whom I have the greatest love, is my grandfather. Without his silent support and the constant encouragement, I would have done nothing during the past year. Along with him, special mentions go to my mother and grandmother.

Considered as family, I would like to thank my girlfriend Tatiana who was always there to love me and support me even when she was suffering from my bad temper. I thank you and I love you.

I would also like to express my sincerely thanks to my friend Maria Stamatopoulou for her support and material provision during the past year, and of course all my friends in Greece who did not forget me when I left, and never stopped contacting me to empathise with my feelings and lift my mood. Finally, I would like to mention one of my former professors who finally became friend; Dr. Kyriakos Kourousis, and thank him for his support. Really, thank you!

Of the people whom I know from Cranfield, I have to say a very big thank you to Mr. Panagiotis Giannakakis, Dr Pavlos Zachos and Mr. Eshati Samir, for their guidance, recommendations and material provision without any profit for themselves. I am very grateful to the library staff for the kindness and help, and a special mention goes to Miss Heather Woodfield for her help and advice, and for time that she spent for me. I am very glad that in the end I can call her a friend.

Last but not least, the most special thank you goes to my supervisor Dr Al Savvaris. I could not have imagined being treated as a friend by someone superior to me. I thank him from the bottom of my heart, and I wish him all the best for his life.

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Nomenclature and Abbreviations

ACARS	Aircraft Communication Addressing and Reporting System
ACMS	Aircraft Condition Monitoring System
ATSU	Air Traffic Service Unit
BITE	Built in Test Equipment
CBM	Condition Based Monitoring
CBMM	Condition Based Monitoring Maintenance
σCF	Centrifugal Forces
CMS	Central Maintenance Computer
CMS	Central Maintenance System
DAR	Digital ACMS Recorder
DMU	Data Management Unit
ECU	Engine Computer Unit
EDMS	Exhaust Distress Monitoring System
EDMS	Exhaust Distress Monitoring System
EIVMU	Engine Interface and Vibration Monitoring Unit
FADEC	Full Authorizes Digital Engine Control
FCM	Fault Coefficient Matrix
h	Height
ICM	Influence Coefficient Matrix
IDMS	Ingested Debris Monitoring System
IDMS	Ingested Debris Monitoring System
ISC	Industry Steering Committee
IVHM	Integrated Vehicle Health Management
JSF PHM	Joint Strike Fighter Prognostic Health Management
LMP	Larson-Miller Parameter
LRUs	Line Replaceable Units

MCDU	Multipurpose Control and Display Units
MDDU	Multipurpose Disk Drive Unit
MEL	Minimum Equipment List
MSG	Maintenance Steering Group
MSI	Maintenance Significant Items
NFF	No Fault Found
OSA-CBM	Open Structure Architecture Condition Based Monitoring
r_{cg}	Centre of Gravity radius
RUL	Remaining Useful Life
SAR	Smart ACMS Recorder
SOT	Stator Outlet Temperature
t	time
tf	time to failure
Tb	Blade Temperature
Tcin	Inlet Coolant Temperature
TET	Turbine Entry Temperature
Tg	Total Gas Temperature
ε	Cooling Effectiveness
εc	Convection Cooling effectiveness
εf	Film Cooling Effectiveness
ρ	Density
φ%	Percentage of Cooling Flow
ω	Angular Velocity of the Roto

Chapter 1: Maintenance Practices

1.1 Introduction

Maintenance should be a major consideration for every company whose operations depend on machines, in order to maximize both quantity and quality of their products. Maintenance actions are defined as those that result in the restoration of a specific component to a serviceable condition or those that prevent the break-down or performance degradation of the component. For this reason the study and development of maintenance techniques is very important.

Apart from adverse weather, air traffic congestion, flight crew and passenger problems, aircraft maintenance is the main factor that impacts on an airline's services because it keeps the aircraft on the ground. Maintenance or technical problems, affect the Dispatch Reliability (DR) of the airline, which shows the percentage of revenue departures which were not interrupted by delays or cancellations due to technical problems. DR, along with the Direct Maintenance Costs (DRCs), which are the costs incurred by the maintenance of an aircraft, are the two values that must be kept high and low respectively in order for the company to make high profits with maximum exploitation of maintenance resources (Knotts, 1999).

A significant improvement in the desired levels of these two factors could be achieved by devising an accurate and specific maintenance plan. Optimum planning is essentially the optimum combination of the different maintenance techniques used. In the following chapter an analysis of those techniques will be presented. These will be divided into maintenance by level and maintenance by time, with its associated subdivisions.

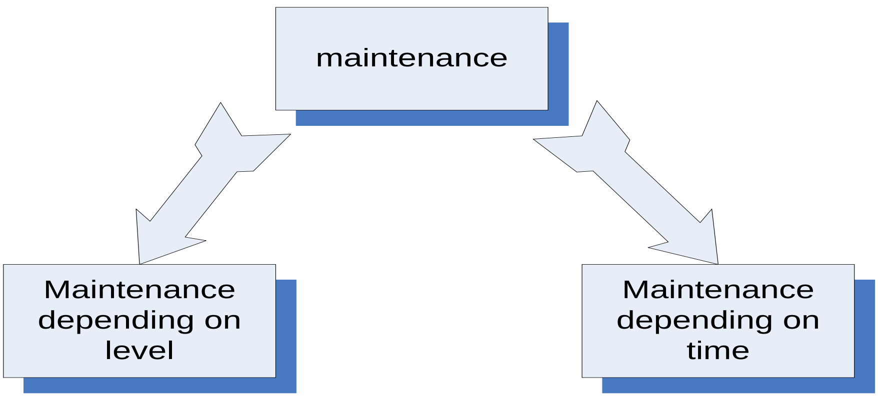

Figure 1-1: General division of Maintenance (Source: Author)

1.2 Maintenance According to Level

The first philosophy of maintenance depends on the depth of the associated maintenance activities and on the level of detail of the maintenance level. There are three categories, each of which refers to specific actions. The differences between these operations are due to the facilities, the tooling equipment and the personnel needed to perform each of them. Additionally, the time taken for each action to be accomplished is another restriction and is essentially the base line of every successive level. However, the separation between one level and another is blurred. Therefore, the following description is as it was found in Airbus internal publications (Airbus Industry, 2010c).

The lowest maintenance level is the Line Maintenance or on-aircraft maintenance. This is performed before the aircraft's next flight and generally concerns actions that will not remove the aircraft from the flight schedule. The content of such a level is restricted to a quick identification and rapid replacement of faulty equipment. Such equipment are known as Line Replaceable Units (LRUs) and are designed so as to be quickly and easily removed and replaced (Kinnison, 2004). In modern aircraft this operation is continuously enhanced as maintenance enhancement systems are established that have the capability to isolate and indicate the faulty systems. A detailed description of such systems will be presented in following chapter.

According to AMC 145.A.10 (European Aviation Safety Agency, 2003), the actions that are included in line maintenance are usually those related to troubleshooting, defect rectification, removal and replacement of equipment and minor repairs or general servicing. A final check is required in order to ensure the proper functioning of the system. The checks and inspections that are performed at this level do not require special tools or in-depth analysis. In general the transit, the 24 - 48 hours, and the A - B check is performed at this stage. On every occasion the aircraft is subject to the preflight inspection at the line facilities.

The second level is the Base Maintenance. Hangar or main base maintenance is characterized by the intervention of maintenance personnel over a longer period of time, and generally concerns actions that cannot be performed at line maintenance level. For this type of maintenance, the aircraft is removed from service, and so there is a relative in convenience of time.

The major activities performed in base maintenance are the highest “C” and “D” Checks, and also any modifications on the aircraft or the aircraft’s systems by service bulletin or engineering order. For that reason, base maintenance activities require more skilled personnel combined with more sophisticated tools and facilities. Finally, the most complex are those inspections which do not rely solely on visual techniques. Non-destructive testing, which require special tools and a longer amount of time can be undertaken. However, activities that could be performed with the modification of certain details can easily be transferred to line maintenance if the operator obtains the relevant license from the manufacturer and the relevant authorities. Activities at this level are supported from the recently established maintenance enhancement systems, such as the Aircraft Condition Monitoring System, in addition to their ability to perform long time trend monitoring.

The final stage is Workshop Maintenance, which refers to both of the previous levels. Due to lack of time involved in Line Maintenance, any faulty LRUs that may have been removed are sent to the workshop in order to be made serviceable again, while equipment or components that have been removed from the aircraft during Base Maintenance are also sent to the workshops for the relevant actions. These types of equipment are easily removed and the maintenance can be carried out off-board. For example, hydraulic actuators or electrical panels can be removed and repaired off-board, unlike the structural repairs, which have to be performed on the aircraft.

These repairs need specialized workshops. Hence, every hangar has workshops which specialize in different aspects of aircraft maintenance, for example the hydraulic workshop, the avionics workshop, etc. Finally, for equipment for which there is no monitoring method or monitoring equipment installed, it is assumed that the workshop maintenance will restore them and find every ‘hidden’ failure. Such systems are those which do not have any Built in Test (BITE) function, or else the test via a control panel located in the cockpit or at the ground is not possible.

1.3 Maintenance According to Time

The division of maintenance according to time is referred to when the maintenance is going to be performed. For an airline, the level of maintenance must be linked with time, so the scheduling and the planning of these activities is essential. However, unforeseen events that require maintenance may occur, hence the schedule cannot be applied. Since safety relies on those maintenance activities they are characterized as unavoidable, and therefore, a maintenance deviation in time takes place.

Although it is sometimes inevitable, it is desirable to avoid maintenance actions that are not time-specific, known as unscheduled maintenance actions. According to Kumar et al. (1999) unscheduled maintenance actions cost a total of £1 million per wide body aircraft for every operational year. To counteract this, airlines demand from the aircraft manufacturer guarantees in order to secure Maintenance Free Operation Periods (MFOPs), during which the maintenance activities will be kept to a minimum. That period is followed by a Maintenance Recovery Period (MRP) in which all the maintenance action could take place, but with the single but very important difference that they will be scheduled (Wu et al., 2004).

Apart from the unscheduled or reactive maintenance mentioned earlier, there is also the category of scheduled or proactive maintenance. However, while this category is correct, it is incomplete and has now been superseded, and the trend is now moving towards maintenance based on condition monitoring (CBMM), which constitutes a new and separate evolutionary category of maintenance known as predictive maintenance. It is difficult to categorize this type of maintenance as being either scheduled or unscheduled. However, because the main interest of this research is CBMM, the complete architecture will be described later, and in this case is assumed to be an element of preventive maintenance, as shown in Figure 2.

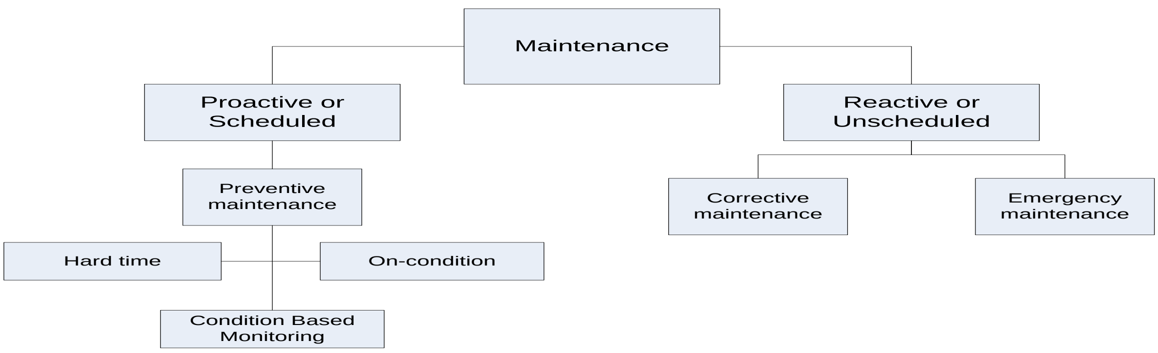

Figure 1-2: Maintenance division according to time (Source: Author)

1.3.1 Reactive or Unscheduled Maintenance

The small but not insignificant category of maintenance according to time is known as reactive or unscheduled maintenance. In general terms, unscheduled procedures fall into two major categories, namely breakdown maintenance and corrective maintenance. In breakdown maintenance the operate-to-failure and the run-to-failure concepts are promoted, whereby a maintenance action is taken only when the component has failed completely and is no longer fit for purpose (Peters, 2006; Beebe, 2004).

Although this methodology has its advantages, such as taking a component's life to its maximum, there are obstacles for its use in the aviation industry and it is subject to serious restrictions due to the fact that safety must come first. Hence, the more appropriate terms for aviation applications are corrective maintenance and emergency maintenance, depending on how urgently these maintenance actions need to be undertaken. The fact that corrective maintenance is performed with no warning, which means that it is unscheduled, makes that method similar to emergency maintenance. The majority of unscheduled activities have an emergency character and they are performed because the safety of the next flight depends on these corrections. However on many occasions unscheduled tasks are the outcome of scheduled tasks. In most cases the tasks included within corrective maintenance are the same as those of preventive maintenance, the only difference being that on the last occasion they would be pre-defined and scheduled.

The definitions that are published for that type of maintenance are numerous, but all are in agreement. Because the area of interest of this research focuses on the aviation industry, the most appropriate definition would be that which is given by the Air Transport Association, and is as follows:

"Corrective maintenance: All the actions performed as a result of failure to restore an item to a satisfactory condition by providing correction of a known or suspect malfunction and/or defect" (ATA, 1992).

The actions that are mentioned in the definition include everything which is done up until the failure is restored and the damaged system is returned to operational use. For many authors these actions begin with the identification and isolation of the defect, followed by whatever action is necessary to restore the system, namely disassembly followed by the replacement or repair of the component, and reassembly. In every case the last action should be the testing of the system.

One accurate and detailed classification of the actions performed within corrective maintenance is given in Dhillon (2006). In this approach, the process from initial identification to restoration of the item to full operation is divided into five steps, each of which includes the tasks that are appropriate for that stage. The steps are as follows:

- **1st Step: Failure Recognition**
  This is the initial step of realizing that there is a failure of a component, system or an item in general. The recognition may come from direct observation of the damaged system or - and most usually - from a cockpit indication combined with degraded performance. In this step lies the true difference between preventive and corrective maintenance, which is that the problem and the failure must exist before every corrective action is performed (Mobley et al., 2008).

- **2nd Step: Failure Localization**
  The outcome of the first step will be failure localization. Even though the recognition of a failure also indicates the location of the failure, the second step does not have that responsibility. The purpose of this step is the investigation of the component which is causing the damage, and additionally, whether the damage to one system has affected another nearby.

- **3rd Step: Diagnosis within the Equipment or Item**
  Following the previous step is the diagnosis of what the failure is and how it can be fixed. This crucial because the correct diagnosis can prevent the problems caused by a wrong diagnosis. The worst case scenario is the combination of unscheduled maintenance with a No Fault Found situation (NFF). According to research conducted for British Airways, the NFF situation cost a total of £20 million, with 8000 removals per month and an average of 14% NFF (Gatland and Trevor, 1993). Therefore, the last three steps are dedicated functions of monitoring systems on modern aircraft.

- **4th Step: Failed Part Replacement or Repair**
  Obviously this is the core of the maintenance activity and includes everything that is done in order for the system to return to service. The main activities performed are the replacement, repair, or servicing of the failed item or equipment. However, the system is not fully operable after that step. The final touch is provided by the 5th step.

- **5th Step: Return System to Service**
  In the aviation industry, great importance is attached to the final check of every piece of equipment, and that is the essence of this final step. Minor alignments and/or adjustments may be necessary, and they will only be apparent after a full check.

The fact that corrective maintenance is an unscheduled event makes it undesirable for the airline operators, because unscheduled downtime brings with it its own consequences. As shown in Dhillon (2006), downtime can be divided into the following categories (Figure 1.2). A similar categorization is presented in Knotts (1999).

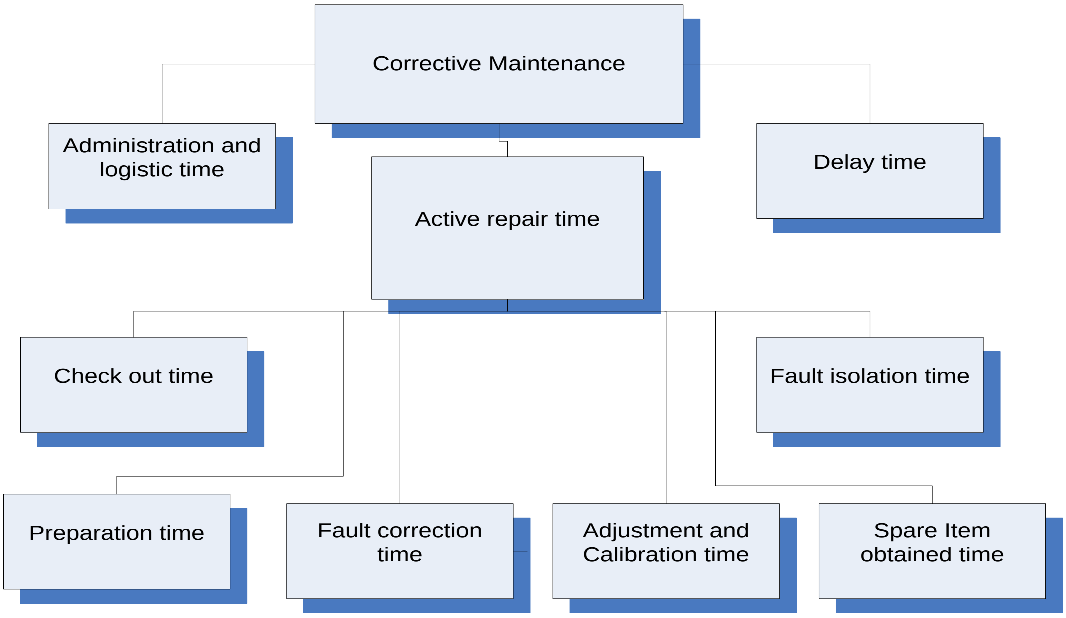

Figure 1-3: Deviation of corrective maintenance downtime (Source: Author)

Since total elimination of these downtimes is impossible, improvement is essential. Taking this into account, and also the fact that maintenance is an unavoidable action for systems/components when they exceed a specific limit, aviation manufacturers are trying to develop methods that can offer, as much as possible, a reduction in these downtimes. Therefore, in every design philosophy that is followed each time, namely Safe Life, Fail Safe and Damage Tolerance, the concept of reliability and maintainability is integrated in order to improve the maintenance process regarding economics, safety, logistics and availability (Wikstén and Johansson, 2006).

The primary goal of maintenance downtime reduction, as studies have shown could be achieved by improving basic factors such as the initial design configuration of the aircraft, but also by making the maintenance process easier and the dispatch permission more flexible. An improvement is been underwent for every of the previous sectors. Initially, the paramount importance should and is given in the design factors that are responsible for the inherent ability of the aircraft to produce failures (Wu et al., 2004).

The first design improvement is the accessibility of the systems and the components. As has been observed from past experience, the majority of the time is taken up in accessing the failed parts. Thus, by paying attention to this at the design stage, a substantial saving can be made. Further savings can be achieved by improved fault recognition, localization, and isolation (Dhillon, 2006). This should be the second design consideration because easy access within the system is not enough on its own. It is hard for a complex system to recognize where the failure is, and thus any help provided by any means is desirable. Additionally, extra costs could be incurred when the recognition is not accurate and a NFF situation arises (Wu et al., 2004).

The improvement of these factors could result in significant improvements for both corrective and preventive maintenance, the difference being that in a scheduled maintenance action, when the operator has to take the aircraft out of service, the downtime is already determined. However as mentioned earlier, improvements could be made by making the dispatch permission more flexible.

Thus, and referring mostly to downtime reduction caused by unscheduled events, the operators, in cooperation with the authorities, have established the Minimum Equipment List (MEL). The purpose of this list is to permit an aircraft to fly safely despite the fact that some systems or pieces of equipment on it are inoperative (Kinnison, 2004; Federal Aviation Administration, 2010). The MEL is unique for every type of aircraft, however the operator can change it if he receives approval from the regulatory authorities and the manufacturer. Finally, the MEL defines the intervals at which the systems or the pieces of equipment that it lists must be rectified (Bristow and Place, 2011).

Additionally, the manufacturers have developed Line Replaceable Units (LRUs), based on the concept that the majority of systems on the aircraft are hosted in boxes that can easily be removed and replaced. Initially, this method was intended to provide maximum improvement for avionics systems, however, it is also being used in conjunction with a number of mechanical systems. The LRUs, along with maintenance enhancement systems had boost the maintenance especially the Line Maintenance procedures which in turn triggers a reduction in Turn Around Time (TAT). Consequently, the operator will reap greater benefits if the aircraft does not use the line facilities of a foreign airport any more than is necessary (Papakostas et al., 2010a).

However, it is the author's opinion that the greatest improvement is likely to be brought about by the full integration of condition monitoring systems, which will assess virtually everything related to maintenance performance. The concept of condition monitoring will also improve deficiencies resulting from human factors, logistic organization and inaccurate maintenance planning orders which do not apply consistently to each different use that an aircraft could have. For example, maintenance planning manuals are created for certain aircraft types even though not every aircraft of the same type is likely to be flying under the same conditions. This is where the true benefits of CBMM will be reaped, and a discussion of these benefits will be presented later in this study in the form of a case study concerning the impact of the take-off performance conditions on the maintenance activities and remaining useful life of an aero engine.

1.3.2 Preventive or Scheduled Maintenance

The term ‘preventive maintenance’ was first introduced in 1957 in a handbook of engineering maintenance. In a rapidly changing world where the accurate management of peripheral maintenance actions such as logistics management was necessary, the disadvantages resulting from the deficiencies of corrective maintenance were no longer affordable. For that reason, and in order to achieve optimum resource organization, a preventive approach to maintenance was established (Stapelberg, 2009).

The fact that all maintenance actions spring from the principle of fixing something before it breaks made the preventive model of maintenance a scheduled model of maintenance. This concept was quickly adopted by the aviation industry because it completely endorsed the concept that safety must come first. This concept was termed Reliability-Centered Maintenance, and was quickly developed in order to achieve cost-effectiveness by minimizing downtimes along with the elimination of chances for failure (Kothamasu et al., 2006). Since 1957 many definitions have been given for preventive maintenance, but it is considered appropriate to reiterate the definition as provided by the Civil Aviation Authority, as follows:

“Preventive maintenance. All actions performed at defined internals to retain an item at a serviceable condition by systematic inspection, detection, replacement of worn out items, adjustment, calibration cleaning etc.” (ATA, 1992).

Even though the actions that are performed in preventive maintenance do not differ from those of corrective maintenance, the fact that they are predetermined provides a very clear picture of what is done and when. According to Dhillon (2006), in order to perform preventive maintenance we have to deal with seven elements, of which the first and most important is the periodical inspection. In this initial operation, the mechanic/inspector must check the desired system, component or item in order to determine its serviceability and whether the item still meets its initial standard. Calibration is the next stage, in which the detection and adjustment of any discrepancy in the component takes place.

In addition to inspection, testing is performed periodically in order to detect any kind of degradation. The testing procedures could be part of the inspection, as noted earlier, but they are also part of the final release. Servicing is the element that includes any periodical actions that should be undertaken in order to avoid incipient failures. These actions could be lubricating and cleaning, and are not to be confused with the installation; that is, the periodic replacement of predetermined limited-life items (European Aviation Safety Agency, 2003).

Finally, adjustment and alignment are the corrective actions which must be done after a service or installation, and include minor changes that improve the performance of the component.

The sequence in which these maintenance elements are listed by Dhillon (2006) is not what actually happens in a real aircraft maintenance performance, and such tasks are the elements of proactive maintenance. Additionally, in a real maintenance operation, some tasks could be omitted if the manufacturer makes it clear in the maintenance manual. This is understood in the context of aircraft maintenance because for almost everything on an aircraft, and more importantly, for the flight critical systems, the manufacturers have conducted extensive tests before launching into mass production. In these tests the systems are subjected to every kind of stress, i.e. mechanical, physical etc., and the results are compared with the optimum values that a system or component has at the beginning of its life.

By means of the usage diagnostics for recommended and expected operating conditions, the manufacturer provides exact and accurate procedures that must be followed by defining the time internals at which they should be done. A similar assessment will be demonstrated later in this study by calculating the remaining useful life for turbine blades. It should be noted that the following case study will concern about the design and not the post service lifing approach. Starting with the inspection and ending with the final delivery in a serviceable condition of the item, the manufactures describes the servicing, installation, alignment, adjustment, calibration and finally the testing procedures.

As established and recognised by The United Kingdom Civil Aviation Authority in 1992 and presented in Knotts (1999) and Kinnison (2004), there are two types of aircraft maintenance under the preventive umbrella. Both depend on the time at which the maintenance is undertaken and they way in which it is organized. Along with condition monitoring, which will receive the greatest interest in this research, and which will be discussed later separately, these three categories also belong to the process-oriented maintenance approach.

The first category and most common approach to aircraft maintenance is known as the hard-time process. Maintenance performed in hard-time means that everything is totally predefined in terms of time as well as process. The systems and the associated components belong to categories that have a definite lifetime, and their use must not exceed a specific total limit. The measurement of exceedence could be counted in various ways such as flight hours, cycles, etc. The maintenance actions, which could be servicing and overhaul, depending on the manufacturer’s guidelines, are performed at the next scheduled maintenance. However, along with the ageing affected components, into this category fall all flight-critical systems and components. For these systems, even though their remaining useful life may be longer, replacement is essential. However, this is the major deficiency of the process; something which is improved by the use of CBMM. Finally, items for which an accurate condition check is not possible, such as composites or plastics, are included in the hard time process.

The mid-point between hard-time and condition monitoring is the on-condition process. The factor that is predefined in this process is the time interval at which a check must be performed, both before and after the maintenance action, depending on the check results. This is because the items that come under that process do not have a definite life expiry, and only a test could establish their current condition. The tests that are performed do not take the form of a simple inspection. They are more detailed and sophisticated tests that ensure the guaranteed and without-failure operation of the system. The tested component must be fully operable until the next pre-defined on-condition test without presenting any in-flight failure. In any other condition, it must follow the prescribed actions such as overhaul and restoration. If a test for a component cannot provide enough information about its condition without performing a tear-down inspection, the component must not be assigned for on-condition maintenance. The components that usually come under this process are the tyres, the brakes and the engine, by performing oil analysis and borescope inspections.

As mentioned earlier, the intervals at which a hard-time or an on-condition maintenance process is undertaken are predetermined and are counted in many ways. For almost every reason, the first time should be when the aircraft is in operation, meaning when the engines are running, known as flight hours. Another option is by measuring flight cycles. One flight cycle begins when the last wheel leaves the ground and ends when the last wheel touches the ground again. However, these time-based intervals are useful for systems that are in operation during the entire duration of the flight or only during takeoff and landing, such as the engines and the hydraulic system, or the tyres and brakes.

Modern aviation uses checks that are a combination of these timed intervals, but also the level of maintenance that will be undertaken. Starting with the Transit Check, the aircraft is subjected to inspection before take-off and after landing. This is the minimum kind of check and the actions that can be taken are limited to line maintenance. The 24 and 48-hour checks follow the same principles. The level of maintenance is again restricted to line maintenance, but the level of the inspection is more detailed. Finally, the most common approach is the check defined by letters. Letter checks tend to eliminate all the other ways of counting and performing maintenance because they combine the hourly counting with the level of maintenance.

Starting with the lighter A Check performed on a regular basis, the aircraft receives low level maintenance. The tasks involved in this kind of check are essentially those of the transit and 24-hour check, and are restricted to visual inspections, servicing procedures and small equipment tests. A more extensive and detailed form of the A Check is the B Check, with its tasks remaining at a low level. This kind of check is usually omitted by airlines and its associated maintenance activities are split between the A Check and the next category which is the C Check. This is a major check and requires special tools, premises and human resources. Specific components are removed from the aircraft for inspection and testing with additional detailed structural inspections. The heaviest version of a C Check is the D Check, which requires the same functions as a C Check, but can keep the aircraft out of service for up to one month (Hessburg, 2000). The time at which a C or a D Check must be performed is determined by the manufacturer, or is planned by the airline after the proposed maintenance planning has been approved by the relative aviation authority. The criteria are not the same for every operator because of the differences which result from the operational environment of the airline, e.g. constant flights over deserts, or the aircraft's operational use, e.g. VIP aircraft do not operate in the same way as a commercial airline fleet.

Every higher check includes every lower check. An airline devises the maintenance planning depending on its needs, thus it adds every maintenance action that must be done in the most appropriate check in terms of time and only. For example, if an A check is close to a C Check, it is possible for actions that have a time window to be performed in the C Check, and to skip the A Check.

1.3.3 Maintenance Steering Group (MSG) 3

In order to create an effective preventive maintenance program it is essential to have available in our hand every kind of availability ranging from data till personnel. This could mean availability of logistics, tools, test instruments, personnel, maintenance manuals and also historical data. However there is a general approach that is covered in six steps, and by following these, an optimum maintenance program can be created (Dhillon, 2006) . The steps are as follows:

- 1st step is the identification and selection of the area that needs extra attention.
- 2nd step is to define the need in terms of preventive maintenance and establish periodic inspection and tasks.
- 3rd step is to determine the frequency of these assignments depending on the manufacturer's orders and the personal experience of the mechanic/engineer.
- 4th step is the preparation of preventive maintenance assignments daily or periodically in an effective manner, and then to get them approved.
- 5th step is to schedule the maintenance activities for a 12-month period.
- 6th step is to expand the preventive maintenance program to other areas on the basis of experience gained from the pilot preventive maintenance projects.

In the context of the aviation industry, these steps could not be applied together efficiently, because in aviation a maintenance programme must be established before the aircraft is launched onto the market. The procedure for creating a maintenance program has been developed over a period of many years before arriving at its present form and terminology. Within the aviation industry, the approach for establishing the maintenance program is determined by the Maintenance Steering Group, with the latest version being version 3 (MSG-3) (Air Transport Association of America, 1970).

This name is derived from the representatives that an MSG program needs in order to be created. These representatives cover the three aspects of aviation; the Regulatory Authority, the Industry Steering Committee (ISC) and finally the Working Groups, each of which has different responsibilities and compositions. These representatives participate as and when required because this is a process which evolves over time, starting from when a new aircraft is ready to go to the market and ending when that aircraft is no longer in operational use (Adams, 2009).

Initially, the ISC consists of the companies who manufacture the aircraft and the airlines that operate it. Even if there is only a single operator, there could be many different manufacturers because different parts of the aircraft come from different manufacturers. The initial maintenance program is created by the members of the group, with the cooperation of Working Groups, and the final decision is approved by the Regulatory Authority. Every final maintenance program is unique for every type of aircraft. If after the establishment of the initial program, a particular operator wants to change it in order to better apply it to their specific program and needs, they can, but once again it must be evaluated and confirmed by the ISC and the Regulatory Authority.

As was mentioned earlier, safety is the first priority in aviation. Thus, the MSG -3 process is a top-down process that organizes the maintenance depending on how a failure on a system or a subsystem might affect the overall safety. This process is accomplished via tree logics with possible answer “YES” or “NO” and their following consequences. The process begins with the manufacturer identifying the Maintenance Significant Items (MSI), which can be either systems or components. The process of identifying them fulfilled in six very important steps:

I. The manufacturer begins to partition the aircraft’s major functional areas as they are defined in the ATA chapters until all the on-aircraft replaceable components have been defined.
II. The manufacturers have to use the top-down approach in order to list the MSI.

III. The following questions must be applied to the MSI:
    a. “Could failure be undetectable or not likely to be detected by the operating crew during normal duties?”
    b. “Could failure affect safety (on the ground or in flight), including safety/emergency systems or equipment?”
    c. “Could failure have a significant operational impact?”
    d. “Could failure have a significant economic impact?”
IV. For any items for which even one question is answered “YES”, the analysis begins, and for those for which all questions are “NO”, no further analysis is required.
V. The possible MSI list is ready and the ISC reviews and approves it, then gives it to the working groups.
VI. The working groups have the last word for the list of MSI.

After the selection of the MSIs, the next thing that must be done is the identification of the 1) function, 2) functional failures, 3) failure effects, 4) failure causes. The analysis is then ready to begin, divided into two levels, and again using the same question answering method. The first level of analysis concerns the consequences of failure. The questions that have to be answered are:

- "Is the occurrence of a functional failure evident to the operating crew during the performance of normal duties?"
- "Does the functional failure or secondary damage resulting from the functional failure have a direct adverse effect on operating safety?"
- "Does the combination of a hidden functional failure and one additional failure of a system related or back-up function have an adverse effect on operating safety?"
- "Does the functional failure have a direct adverse effect on operating capability?"

The second level is the task development level, for which six questions must be answered.

- "Is a lubrication or servicing task applicable and effective?"
- "Is a check to verify operation applicable and effective?"
- "Is an inspection or functional check to detect degradation of function applicable and effective?"
- "Is a restoration task to reduce failure rate applicable and effective?"
- "Is a discard task to avoid failures or to reduce the failure rate applicable and effective?"
- "Is there a task or combination of tasks applicable and effective?"

The evaluation of the above gives the decision tree diagram (Figure 1.4) and creates the Maintenance Report Board (MRB). In addition to the MRB, but always based upon it, the manufacturer publishes their own Maintenance Planning Document (MPD). This is a generic term, and each manufacturer can use their own preferred name; for example, Airbus calls it the Maintenance Planning Document, while Boeing uses the term Maintenance Planning Data.

An MSG-3 maintenance program provides maximum aircraft availability as a result of equipment availability and reduction of extended inspection intervals. It also ensures the aircraft's inherent safety and reliability along with the operational safety, suitability and effectiveness. In addition, it is a cost avoidance method. To summarize, it integrates all levels of maintenance activity while ensuring that all equipment is thoroughly covered by the proper level of inspection (Voith Industrial Services, 2010). Thus, it ensures that the majority of aviation demands are covered. These demands can be simply and accurately summarized in two points; safety must come first, and then after there is no point in maintenance if failure is cheaper (Bristow and Place, 2011; Mann et al., 1995).

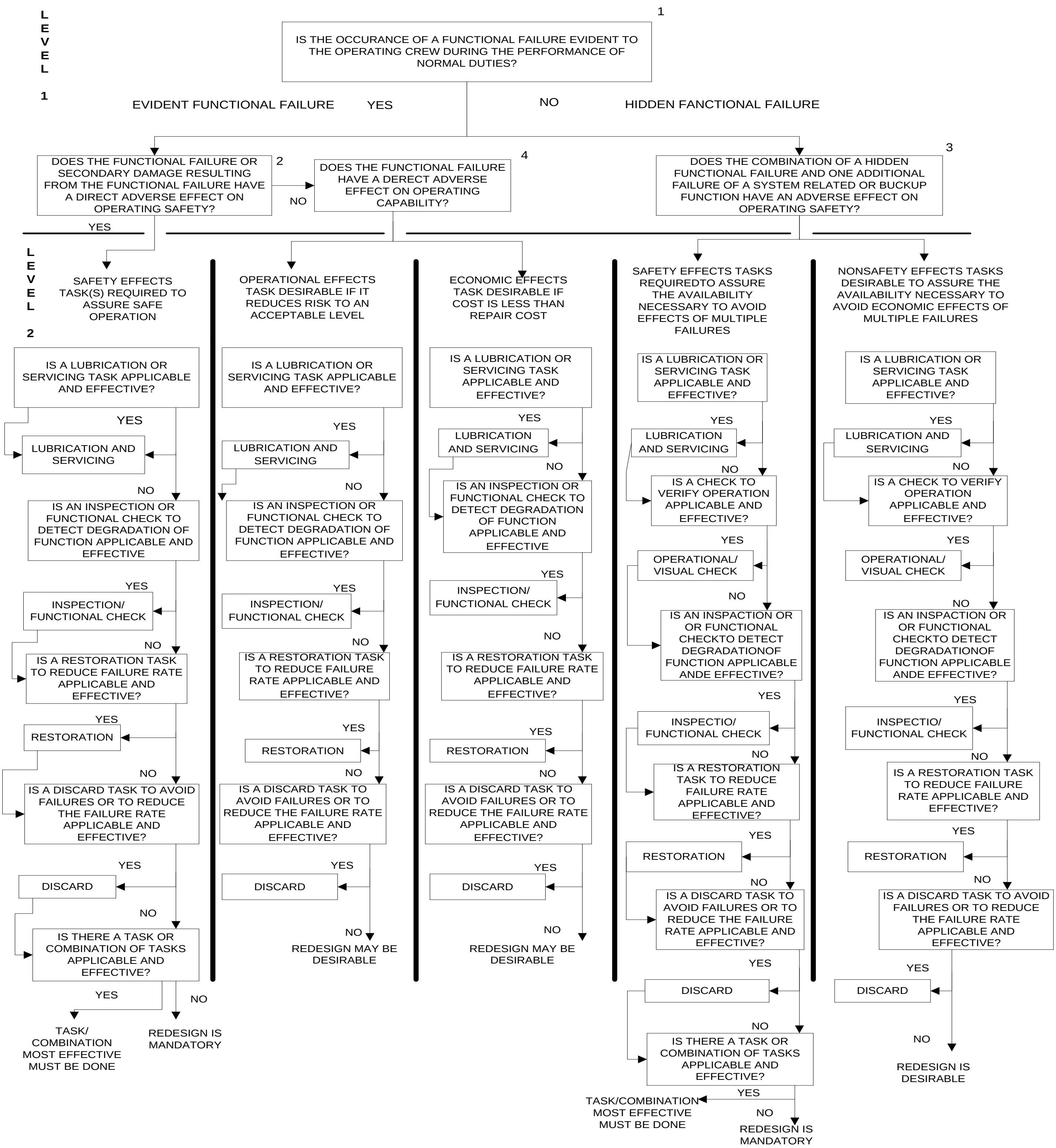

Figure 1-4: MSG decision tree diagram (Source: Author)

1.4 Condition Based Monitoring

Condition Based Monitoring (CBM) is a state-of-the-art technique in performing maintenance which is the called Condition Based Monitoring Maintenance (CBMM). It combines reactive and preventive maintenance, the outcome of which is the predictive way that maintains the values of aviation reliability-centered maintenance (Figure 1.5). Briefly, by using this method, the operator can monitor the state of the component and then schedule the maintenance which is needed at the time, while also exploiting the whole remaining useful life of the component. This method was first developed by Boeing in order to create a new method of preventative maintenance for the new - in those days - 747 model (Bengtsson, 2004), which would respect the reliability-centered concept of the existing MSG, but would be more efficient (Tsang, 1995). The actual research did not refer to CBM from the beginning, but to the tendency of the aircraft components to failure.

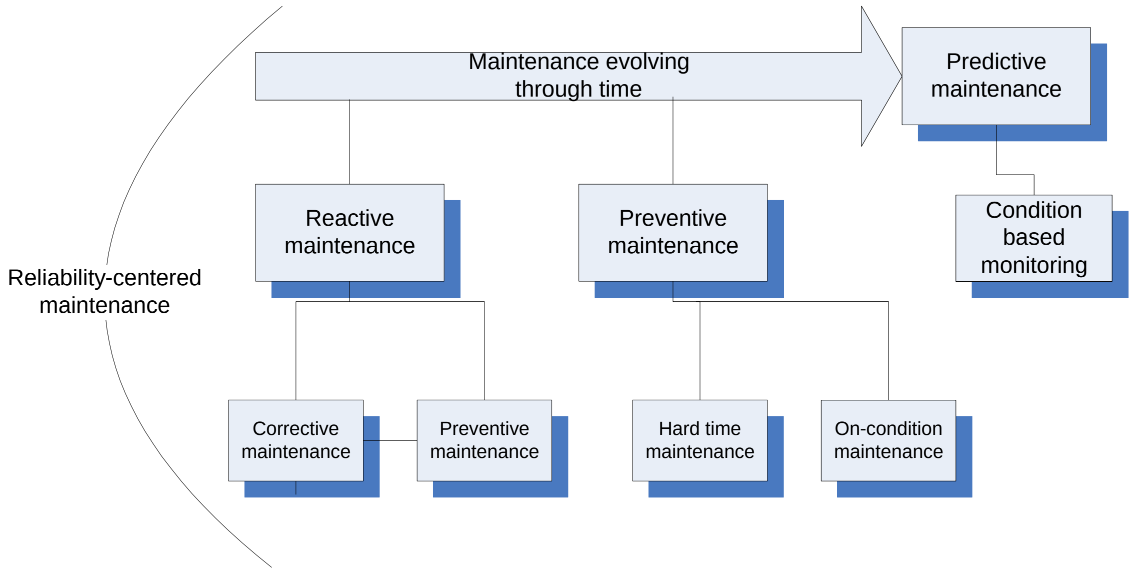

Figure 1-5: Modern approach of maintenance division (Source: Author)

That research was commissioned by the US Department of Defense and revealed that approximately 11% of the components had characteristics that deteriorated as a result of ageing. Regarding these components, the scheduled hard-time maintenance process was a solution which could be defined and predetermined in terms of time or other criteria such as flight cycles. For the remaining 89% which did not show ageing characteristics and whose maintenance could not be ‘predicted’, the need to develop a new method for maintenance was obvious (Nolan and Heap, 1978). That new method turned out to be condition-based monitoring, which provides the opportunity for on-demand maintenance.

In its early stages, this method did not have a clear character, and belonged to both of the major categories of preventative maintenance, namely scheduled and unscheduled. More accurately, due to the lack of monitoring techniques, condition monitoring was restricted to components that did not have a definite lifetime, and as such, their replacement was on demand. The monitoring technique was restricted to the experience and the physical senses of the mechanic, in combination with crew reports referring to system malfunctions.

Most importantly, the components that fall under condition monitoring should follow the specifications that have been defined by the aviation authorities (Air Transport Association of America, 1970; Civil Aviation Authority, 1997). Specifically, the Condition Monitored (CM) item must not be a vital component on which the airworthiness, reliability or safety of the flight depends. Moreover, no ‘hidden functions’, i.e. affecting side components or systems are acceptable if a failure on the monitored component occurs. In addition, the operator has to include the CM items into a data collection and analysis program in order to better understand the reasons for failure, for future use. However, condition monitoring maintenance tends now to be a fully predictive maintenance, something that is achieved by using very sophisticated monitoring, diagnostic/prognostic methods in all of the systems, regardless of their criticality. This is also the difference between on-condition and condition monitoring maintenance.

In recent times, companies are facing a dramatic change because of the need for high productivity in order to be competitive and leading-edge. The key for high productivity is to achieve high levels of availability (Butcher, 2000; Olsson et al., 2004). For an airline this would translate into the maximum availability of their fleet in order to have a maximum operational profile, which means maximum productivity (Papakostas et al., 2010b).

By performing predictive maintenance by using the capabilities of CBM, the operators can benefit from the fact that every component that is been replaced has reached the end of its life. They can avoid unscheduled maintenance actions, and can create an accurate preventive maintenance plan that precisely suits the aircraft’s use. Additionally, the situation of keeping the aircraft on the ground for long periods of time because of a failure in a component that is too big and too expensive to be kept as a spare can be eliminated (Provan, 2003), because the logistics management is improved by predicting the incipient failure.The benefits of utilizing CBM were first observed in the army, which began research in order to implement CBM techniques in existing helicopters and aircraft (Hess, 2002). Currently, a very sophisticated outcome of that research is in use with the state-of-the-art CBM on the JSF F-35.

1.5 Condition Based Monitoring Architecture

In order for a tool such as condition monitoring to be fully operable and for its integration to be achievable, the existence of an architecture capable of hosting its functions is essential. The need for a standard CBM architecture was soon recognized, and a fundamental framework was established (Discenzo et al.). This framework consisted of functional system elements, interface requirements, and also the operational procedures that a CBM architecture, and subsequently a specific CBM system, must implement. This initiative was undertaken by representatives from industries and universities to army representatives, covering everything that is involved in a CBM system (Lebold et al., 2003).

Essentially, what was actually needed from a CBM method was a reduction of the costs of maintenance combined with maximum safety. The safety aspect had already been fully accomplished with the previous techniques, but the losses incurred by the unnecessary replacement of equipment was no more affordable. Consequently the most important requirement of a CBM system is to be able to diagnose the current condition and predict incipient failures for the component under inspection. By definition, that process will show the remaining useful life of the component.

In order to accomplish this central requirement of diagnosing/predicting, the systems have to have separate dedicated functions such as data acquisition and processing, and so the outcome and the combination of these should provide the desired data. More specifically, the key utilities and activities that a complete CBM system should have begin with the data sensing and acquisition, followed by the initial processing of these, along with feature extraction. The result of the data processing is the triggering of the alert mechanism, firstly with the diagnosis, and then with prognostic enhancement.

Additional decision support is desirable, along with possible recommendations in general, and is almost essential for a system to be as near perfect as possible. The partition of every task into separate functions should to be a separate piece of the system, or even better, a separate layer. By setting these requirements as goals for the system, the desired operational requirements and system elements respectively have been covered (Discenzo et al.).

The aforementioned CBM architecture was standardized to Open Structure Architecture-Condition Based Monitoring (OSA-CBM) in 2001. The program began from zero because no framework existed and was led both from industry and the Army through a Dual Use Science and Technology (DUST) program. The participants covered a range of industrial, commercial, and military applications of CBM technology: Boeing, Caterpillar, Rockwell Automation, Rockwell Science Center, Newport News Shipbuilding, and Oceana Sensor Technologies. Other team contributors included the Penn State University / Applied Research Laboratory and MIMOSA (Machinery Information Management Open Standards Alliance) (Thurston, 2001).

During that time several versions of an OSA CBM architecture were introduced, with the latest version, V.3.3.0, taking the following form (Penn State University et al., 2006). Extensive details of all of them are publicly available on the MIMOSA website www.mimosa.org. The things that have been changed from the first version do not apply to the actual layers but to the interaction between them. Furthermore, the human interface, which is considered to be the top layer, does not appear in the latest edition, but that does not mean that it isn't present.

The sequence in which the layers appear refers to the representation of the data flow and to the way in which the elements should be follow one another. It should be noted that the architecture described below was found in many publications. However it seems better to analyze the latest adding some details found in other publications. Beginning with data acquisition and ending with the human display, the seven layers are as follows:

1st lLayer: Data Acquisition or Sensor Module

The first layer is responsible for feeding the CBM system with the correct data that the system needs in order to proceed with the processing and provide results. The sources of these data are sensors that monitor the desired component. Fraden (2003) defines a sensor as a "device that receives and responds to a signal or stimulus". After acquisition these data must always be digitized, but a Word output is also acceptable. The establishment of Micro Electromechanical Systems in the CBM market has reduced the size and the price, and has brought along with it advantages such as temperature stability, self-test functions, low drift and sensor redundancy (Bengtsson, 2004). In the event that the sensor is unable to produce digital outputs from the beginning, a transducer is fitted to complete the task. Essentially, this is a conversion from analog to digital, achieved by adding electrical or optical energy to the signals. In addition, the sensor layer should be able to calibrate the data obtained and must also be able to provide secondary information to its interfaces, such as sampling rate and frequency. In every case a sensor or a sensor layer cannot stand on its own in a CBM system (Olsson et al., 2004). However, it important to pay special attention to that layer, because no matter how sophisticated the diagnostic method might be, they all suffer from sensor noise and bias problems; something that in many cases can compromise the fidelity and the accuracy of diagnosis of the system.

2nd Layer: Data Manipulation or Signal Processing

The data acquisition from the first layer is succeeded by the primary and initial data processing, which takes place in a completely separate layer, with its own dedicated functions. The actual outcome of these processes is the configuration of the data in the form desired by the operator. Normally a Fast Fourier Transform is required in order for the signals to be usable by the other layers. This action can be performed in a single or a multi-channel signal transformation. Before the Fast Fourier Transformation, the signal processing layer is responsible for removing any falsifications or distortions that may exist in the signal and restore it to its original form, but also for removing from the sensors any data which is irrelevant or unnecessary for diagnosis or prognosis (Olsson et al., 2004). Moreover, outputs that contain average values of sampling, standard time deviations, wavelets, frequency spectra and specific extracted CBM features are produced by that layer. Finally, the signal processing layer performs sensor data digital filtering.

3rd Layer: Condition Monitoring or State Detection

An initial judgment about the condition of the monitored component is established by the condition monitoring layer. Operating as a ‘virtual’ indicator, it provides information about the current condition of the component. In order for that to be achieved, a comparison is made between the data and the values received at that exact moment with the data and values that the operator has already predetermined for the system. This means that the system/layer is already fed with a library of optimal and expected data that define the operational limits, and any divergence or deflection from them will trigger the alarm mechanism. It should be noted that there are no assessments of the health of the component in that layer, but only indications that come from the data comparison. The inputs for that layer may come from either of the layers mentioned above. In addition, it commands, coordinates and manages them in order to achieve scheduled and/or on-demand data sampling and processing, which are the responsibilities of the first two layers (Thurston and Lebold, 2001).

4th Layer: Health Assessment

After the initial judgment has been made, the next step is taken by the health assessment layer, whose role is to make assessments and determinations about the current health condition of a monitored component or system, and possibly if there is any degradation, why it has happened. While the inputs are taken from the previous layer, the system/layer is again pre-informed about general issues. The data that the system/layer needs in order to make its assessments refer to the health and maintenance history and the operational status of the component. As detailed above, these historical data are accurate and the system will generate reliable indications. These recommendations include possible evolving faults and diagnostics. After every output generation the system should be able to create a record for future use. The diagnostic processing relies on three different methodologies, each of which uses a different diagnostic method.

The first of these is the rule-based diagnostic system, which is a combination of pre-established knowledge and a set of rules devised by experts in the field (Olsson et al., 2004). Constant updating of this set is essential in order to ensure the validity of the outcomes from the system. A faulty rule could damage the whole diagnosis, which is a major disadvantage for this system. The second system is the case-based diagnostic system, which is based on a library of predefined cases (Chiu et al., 2004). When a fault occurs, the system compares the current fault with the closest and most similar case and provides a diagnosis. The new outcome will be added to the library after human or system verification and revision. The case-based reasoning method is a good solution when the first method encounters difficulties in creating a large rule base. The third system is the model-based diagnostic system, which is similar to the case-based diagnostic system (Mann et al., 1995). This utilizes a manually designed model for the monitored system and any deviation from it will trigger fault detection. Furthermore, these deviations are used in order to define what the fault is. This is the most desirable method, but it is restricted because of the difficulties and the limitations that arise when manually creating a model for the monitored system (Mathur, 2002; Roemer et al., 2001). However, even though all of these systems are separate entities, a combination of them is practical in use. This is because many of the diagnostic methods could be integrated in the system in order to be as complete as possible. The system is further enhanced when artificial intelligence diagnostics applications such as artificial neural networks are used.

5th Layer: Prognostics

Along with the Health Assessment layer, the Prognostic layer attracts the largest proportion of the research that is currently being developed in this field. The function of this layer is to predict future conditions. These predictions take the form of the estimation of the remaining useful life (RUL), along with the current state of the component or system which is being monitored. In addition, prediction of when a failure will happen is desirable. A database which includes the future usage and operational profiles is essential in order for the RUL to be as accurate as possible. The inputs to this layer could come from every layer of such an architecture and could provide information about health, failure, mission and maintenance history (Thurston and Lebold, 2001).

6th Layer: Decision Support

By amalgamating the data that has been calculated, recommended and presented by the previous layers, the role of the decision support layer is to finalize all the assessments that have been made. By utilizing past present and future predicted maintenance and operational data, it is able to provide recommendations for the correct and proper actions that should be undertaken from that point on. These actions range from suggestions for the next scheduled maintenance actions to suggested flight operations. The purpose of the latter is to mitigate potential dangers, in the event that the monitored component or system is about to fail. Also important for the system is the option the human operator overriding the system's recommendations. No matter how sophisticated the decision support layer, the best final decision will come from a human. The faults that a CBM system could detect on the monitored item may be too large or too expensive and complex to correct and so the decision support layer may be inaccurate, thus a final human decision is essential in order to avoid a potentially bad consequence produced by a fully automated system (Olsson et al., 2004). The OSA-CBM does not set restricted and specific standards for this layer, and so a future logistic management enhancement would be possible.

7th Layer: Human Interface or Presentation

The final layer provides the human operator with an operational window. This window is usually a computer screen. In every case the presentation layer shows the results from the six previous layers. In addition, the operator is able to control and interrogate every layer separately and take data directly from the layer that produces or processes them. The most common layers for interrogation are the diagnostic and prognostic layers, along with the alarms/alerts that come from the condition monitoring layer. The data that an operator could extract are not necessarily limited to CBM use.

The system architecture covers all of the interface requirements, with the exception of functional system elements and operational procedures. Interface requirements comprise the communication that should take place between every layer, and is divided into the following categories, according to the type of communication that the data can use to flow into the system, which could be termed data flow communication types. There are four data flow communication types, namely synchronous, asynchronous, service and finally subscription (Figure 1.6), and the module interfaces of each is defined in OSA-CBM UML.

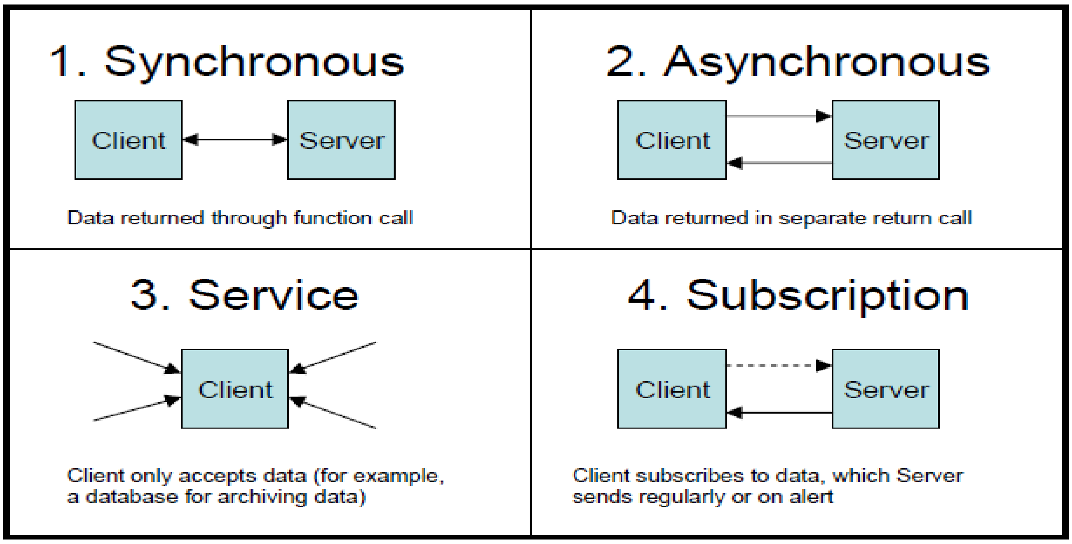

Figure 1-6: OSA-CBM Communication Types (Penn State University et al., 2006; Thurston and Lebold, 2001)

Whichever one of these types is in use, it must be compatible with a variety of data reports and data processing types. There are two categories of data reporting and processing; time based and event based, with the first divided into periodic and aperiodic. Time based reporting and processing takes place at predetermined fixed time intervals, either periodical or aperiodical, when a human commands a report. Conversely, event based reporting and processing takes place when an abnormal and undesired state, such as a limit exceedence arises, which triggers the reporting and processing mechanism. The timeliness of the data reporting and processing also refer to how critical is their report and processing, depending on time. For this reason they are divided into time-critical and not time-critical messages. The Central Maintenance System, described later, which belongs to the CBM systems, follows these principles

As well as the way in which the data flows, the system also needs a communication platform on which these data will travel. This task is assigned to the middleware programming language. Essentially this is the software that enables the dialog and communication in general for separate modules. Examples of such technology are the Component Object Request Broker Architecture (COBRA), developed by the Object Management Group; the Distributed Component Object Model (DCOM), developed by Microsoft; the web-based Remote Procedure Call (RPC) and the JAVA Remote Method Invocation (RMI), developed by SUN.

The standardization of the architecture in the layers that have been discussed earlier can result in several benefits, each of great value. As the communication standard is already set for each layer, each can be upgraded separately, simply by following the predetermined standards. In general, CBM uses a lot of Commercial Off The Shelf (COTS) hardware and software, and so there will be an initial benefit of cost reduction because no additional time is required for developing new architectures. A cost benefit could also result from the user's point of view because there is no commitment to a single seller for a whole CBM system, or even for a specific functional module. Furthermore, the partition resulted in specialization, and now every vendor focuses on a single aspect of the system. More sophisticated algorithms and more accurate technologies should now be developed as a result of that specialization. Moreover, competition between the vendors, at a layer rather than a system level, is encouraged because the user has more than one choice. Conversely, this could also bring about cooperation, with its own inherent benefits.

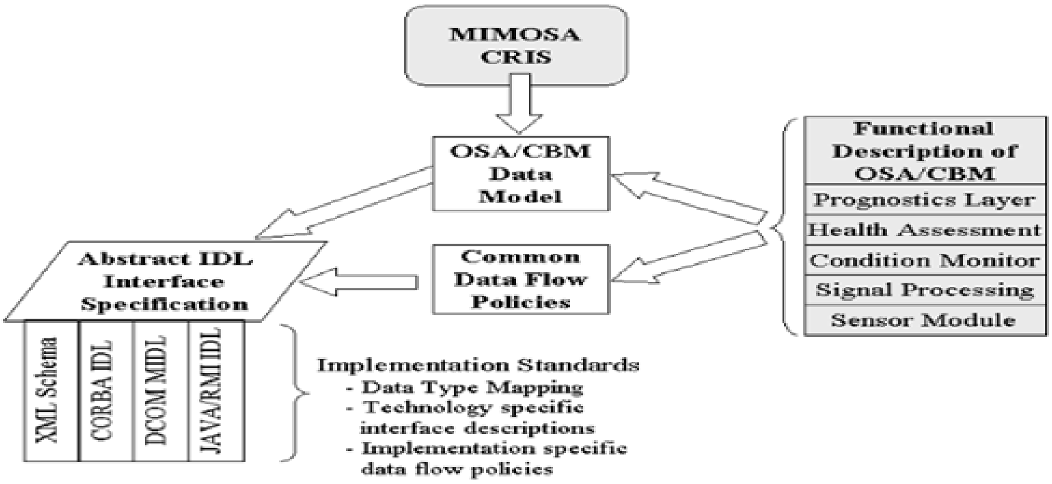

Figure 1-7: Outline of the OSA-CBM Architecture (Thurston and Lebold, 2001)

1.6 CBM Implementations

The examples discussed illustrate that the implementations of condition monitoring systems as stand-alone systems are numerous. However, the trend is moving towards and thus the capabilities that a condition monitoring system provides are being used as the core of a large and very sophisticated maintenance management system. Such systems are known as health management systems, and they cover the entire range of operations involved in the maintenance of an aircraft.

The final objective of a modern health management system is to achieve maximum fleet availability and optimum safety and airworthiness, combined with maximum degradation of costs resulting from factors such as maintenance and logistics. Depending on the manufacturer and the participants, various terms can be used to describe the implementation of health management systems and methods. A number of examples are in use under the generic name of Integrated Vehicle Health Management (IVHM), mainly referring to Boeing and the US Navy, although that doesn't mean that there are no other names for similar applications. Like all of these systems, IVHM uses the OSA-CBM architecture and a complete system is able to incorporate the philosophy, methodology and the processes required to improve safety, operability, maintainability, reliability, and finally testing (Keller et al., 2001; Dunsdon and Harrington, 2008).

The desired objective is achieved by dividing the related processes into either on-board or off-board, each having separate but also interacting functions. Every on-board architecture is able to assess the component or system health for both operational and maintenance purposes. The on-board functions are grouped under the term Operational Maintenance Program.

Examples of on-board capabilities can be found across many military and civil uses. The source of data is always on-board sensors. The primary processing of these data takes place on board in order to fulfil more important needs, while the aircraft is still in flight. These needs are expressed by means of warning signals that the pilot sees, which inform him of a malfunction in one or more systems. These signals are also indications that a maintenance action has to be done when the aircraft lands (Keller et al., 2001).

By adding new technology systems these sensors are capable of receiving signals for multiple uses. In practice, after the sensor module, the systems that receive these data can use them for two purposes. The first, as previously mentioned, is for on-board warnings and alerts, while the second is for performing long term trend analysis of the monitored system or equipment, which is equally useful for maintenance purposes. Both types of data can be stored in on-board memories and also be transmitted to the ground, depending on their importance.

The health management systems available for commercial aircraft illustrate their rapid evolution since the time that the first on-board alerting systems were introduced. These first generation systems used light bulbs which lit up when an interruption of signal or power was observed. Their evolution began with the second generation systems that were first introduced by Boeing and Rockwell-Collins and Airbus on the 757/767 and A320 respectively, and with the integration of built-in test equipment, whose responsibility it was to detect anomalies on their dedicated system, the process was simplified considerably. All the warnings that were generated were collected from the Maintenance Control and Display Panel (MCDP), for Boeing aircraft, and from the Centralized Fault Display System (CFDS) for Airbus. Due to the fact that the CFDS was introduced later, it has a very sophisticated capability (for those times) to translate the cryptic codes into English text, and thus much decoding time was saved.

In the early nineties the establishment of the Central Maintenance Computers (CMC) or On-board Maintenance System (OMS) gave a true enhancement to the mechanics because these systems were not only able to indicate which LRU was faulty, but also which maintenance procedures were required. With continued improvement in the CMC concept, the 777 and A340 are equipped with the next step of the initial CMC. An extensive analysis of maintenance enhancements systems follows in Chapter 3. Improved diagnostic capabilities using model-based techniques are now available on these systems. Additional help is now provided by portable devices that enable the mechanic to be as close as possible to the root of failure, without needing to return again and again to the cockpit (Bird et al., 2005).

In terms of maintenance activities, the on-board systems are capable of diagnosing the failure and provide significant enhancement to the troubleshooting process. However the sampling undertaken by the on-board systems generates the off-board process, wherein lies the true evolution of health management systems. By performing trend analyses, the off-board devices are capable of predicting incipient failures and planning the maintenance activities according to when these failures are likely to happen. The fact that prognostics capabilities are very sophisticated keeps unexpected failures to a minimum, such that they only arise as a result of some external event, such as a bird hitting the engine. The data can be downloaded either via transmission from the aircraft to the ground or by extracting them from the memory modules located on the aircraft via portable ground systems.

The IVHM architecture incorporates two modules within the off-board domain. The first is the Maintenance Data Warehouse, where all the information is collected and stored. The loaded data are processed by a Ground Based Reasoner which produces diagnostic/prognostic data which is fed into the Maintenance Reasoner. The latter supports the maintenance personnel by indicating precisely which component needs to be repaired. In addition, the process is completed by means of a Dynamic Resource Manager function, which takes account of the availability of resources, but also of the future operational profile of the aircraft, and according to these, indicates the optimum time at which to perform the maintenance activity.

The on-board/off-board partition architecture provided by IVHM is in use in several ongoing applications, which include the Navy/Boeing Dual Use Reconfigurable Control and Fault Identification and OSA/CBM programs, as well as the DARPA/USAF Unmanned Combat Air Vehicle (UCAV) program, with the latter targeting the fuel system of the aircraft. In addition, a huge effort has been made for the helicopter Health Usage Monitoring System (HUMS) in order to predict incipient failures on intermediate gear boxes, while under the Dual Use Science & Technology program (DUST) the US Navy is collaborating with General Electric for the Integrated Engine Prognostics and Health Management system (IEPHM), for aircraft fitted with GE engines. General Electric is also collaborating with Boeing for the Aircraft Electrical Power System Prognostic Health Management (AEPHM) (Butcher, 2000; Hess, 2002; Dunsdon and Harrington, 2008).

The state-of-the-art in health management programs, however, is being implemented on the modern JSF F-35 fighter jet, with its own dedicated program, Joint Strike Fighter Prognostic Health Management (PHM). This is being developed as a collaboration between Lockheed Martin, Boeing Aerospace, Pratt and Whitney and General Electric. The complete system is able to reduce life cycle costs, and consequently the maintenance man hours needed per flight hour. The key features of this program are the advanced technologies that are implemented both on-board and off-board.

The on-board system is able to perform fault detection and fault isolation with a very high level of accuracy, and also provides very accurate prognostic data. Every subsystem on the aircraft is equipped with a dedicated PHM Area Management system, all of which are connected to the central Air Vehicle PHM Manager. The flight critical events are processed in the air and only the most necessary operations are left to the pilot. The non-flight critical events are used for long term analysis on the ground. The off-board systems take account of all the operational usage, not only of the aircraft, but of the whole fleet as well. Finally, the connection of the system with the Autonomic Logistic Information System (ALIS) provides the extra advantage for the operational usage of the aircraft (Brown et al., 2007; Ferrell, 2000).

To summarize, the improvements and benefits that result from a condition based monitoring maintenance methodology were quickly recognized by the industry. From the point of view of the US Army CBM team, this technique offers many advantages. First and foremost, due to the lack of unscheduled corrective maintenance actions, the availability and airworthiness of the fleet are improved significantly, thus addressing the two major dimensions of aircraft maintenance. Firstly, safety is not compromised because the flight critical systems are monitored, and secondly, thanks to the prognostic capability, no unexpected problems occur, thus unscheduled downtime is eliminated. The whole procedure also helps to improve the troubleshooting process, which increases the confidence of the crew by making them feel safer. Moreover, the economic advantages are also considerable, as savings are accrued by eliminating the downtime throughout the life of each component.

Chapter 2: Engine Diagnostics

2.1 Introduction

The concept of condition based maintenance, along with all the benefits that it carries, requires the existence of monitoring and diagnostic methods that will apply on the previously described architecture. For the present study, the area of concern is jet engines as a part of the whole aircraft. Thus, a description of existing monitoring/diagnostic methods will be presented in this chapter as part of this research, and before any of the results coming from one of them will be demonstrated. It is the author's belief that a description of these methods will help the reader to have a complete view of the technologies in use for condition based monitoring, and to understand what lies behind the diagnostic systems that follow in the next chapter.

Even if the engine is concerned to be a part of the aircraft, it is divided into subsystems which it is essential to be aware of, in order to know where the monitoring/diagnostic methods are being applied. The basic arrangement of a turbine engine is divided into three parts: the thermodynamic elements, the rotational mechanical elements, and the accessory equipment. For a complete and effective engine health monitoring system, the integration of techniques that cover the whole spectrum of the components is essential. It should be noted here that the majority of the diagnostics refer to post service assessments which means that they provide results when the system is already in use. However some of them like the usage diagnostics could be used for estimation from the design stage of the engine. Such an example will be presented later in the case study.

The development of technologies to facilitate estimations about the remaining useful life and the fault detection of the engines' components has been under way for a number of years, and so some core monitoring/diagnostic techniques have been used in practice for a long time. However, the requirement of a health management system that is capable of providing diagnostics for the engines has led to the creation of new and more sophisticated technologies, and with greater capabilities, although that by no means suggests that the old techniques are redundant.

These state of the art diagnostic systems provide results by utilizing all the available monitoring technologies. Specifically, there are mechanical diagnostic methods, which in general refer to the old monitoring techniques and apply to a specific component. They lack complexity, however, with the associated advantages. On the other hand, there are performance diagnostic methods, which have the advantage of assessing the life of the whole engine and its components by utilizing a model of performance data for processing. Finally, the state-of-the-art diagnostic methods are those which utilize artificial intelligence, and combine characteristics from the previous two.

2.2 Mechanical Methods

2.2.1 Oil Analysis

Oil is a necessary component in an aero engine, partly as a lubricant to reduce the friction between the countless rotating parts, but also as a coolant, as it reduces the temperatures caused by such friction. Observation suggests that changes in the properties of the oil, such as consumption, or the appearance of combustion derivatives, could result in a malfunction in the parts with which it comes into contact. Hence, several methods have been developed for providing engine diagnostics by examining the oil within them. It is been proved that condition monitoring of the oil is a very reliable primary technique for engine diagnostics, and for this reason it is assigned to the engine diagnostic methods and is more accurate than the mechanical ones.

The main focus of oil monitoring is to detect any changes in the oil's physical and chemical properties, or to detect particles within it. The results of an oil analysis, or preferably the combination of that analysis with other monitoring methods, could produce accurate and reliable indications of the condition of the component, and hence could constitute a significant enhancement to the predictive maintenance process.

Oil analysis-based diagnostics share two great advantages, namely low cost and lack of complexity in being performed. However the core techniques of oil analysis give reliable but basic, unsophisticated results, and so it is necessary to implement more sophisticated technologies in order to achieve the diagnostic goal. However, depending on the time window to which the analysis applies, oil analysis is separated into on-line and off-line processes, which generally refers to the processes that can be performed on the aircraft (on-line) and those which have to be performed on the ground (off-line), but which concern the same component.

The on-line process refers to those methods that are capable of providing results at the exact time of measuring and sampling, and requires constant monitoring by the system. Modern systems use the oil debris detection method in order to provide instant indications of faults. However, systems that indicate the oil temperature or oil consumption and are hosted as standard on an aircraft could also be classified as on-line process systems, even if they don't have a direct diagnostic purpose (Hörl and Richter, 1995). Conversely, the off-line process is characterized by the convenience of time, and mostly applies to methods that require laboratory chemical analysis. Moreover, it is possible to perform more detailed analysis that requires past and present record data for prognostics reasons.

Different techniques have been applied over the years for oil analysis, each focusing on a different aspect, with the fidelity of the results being affected positively by the combination of methods. However, it is clear that these oil analysis methods can be categorized into two general groups. The first group refers to those techniques that focus on finding any physical or chemical changes within the oil itself. This group is known as Oil Condition Monitoring, and includes the three most commonly found analysis techniques.

The first and simplest technique is the Blotting Paper Test. This is a simple visual method which uses a special paper which takes on different colours depending on the clarity and the condition of the oil. Next is the Capacitance Test, a method that is used in order to determine the contamination of the oil resulting from static electric charges, the presence of water, methane debris and acidic oxidization. The presence of any of these will result in a reduction in the oil's capacitance, and so provides an indication of an incipient failure. The final method is the Viscosity Test, which is carried out in the laboratory. This test usually focuses on fuel contamination, which is indicated by a decrease in viscosity, which may mean fuel dilution (Li, 2008).

The second group of methods is called Oil Debris Monitoring. The purpose of such techniques is the detection of contaminant debris found in the oil. While such debris could take the form of solid particles, or else could be liquid or gaseous, this method focuses primarily on solid particles because they could be an indication of wear on engine parts, especially on bearings. The amount of the debris found provides indications of the condition of the bearings, and most importantly for this case, whether they are degraded. It should be noted that not all solid particles are the outcome of the process of wear, as some could be ingested from the atmosphere. Additional attention should be paid to the time at which such debris is produced, because there is an acceptable amount of wear for all parts when they are new and have just been installed (Miller and Kitaljevich, 2000).

For this reason modern oil debris monitoring systems combine the results of the analysis with model-based knowledge which is predefined cases in a model form in order to provide assessments of the remaining useful life of the component they refer to (Orsagh et al., 2003). Because debris monitoring focuses on the debris resulting from bearing wear, and because the engine bearings are considered to be flight critical parts, a significant amount of development has gone into these techniques and they can now be found on many engine applications as an on-line process system. The most common methods for detecting debris are ferrography, the magnetic chip detector or magnetic plugs, and analysis of the filter for debris by using microscopes (Rao, 1996).

One state-of-the-art oil monitoring technique is being developed under the JSF PHM program. With this technique, the engine is equipped with a system ready to provide on-line information about the condition of the engine by exploiting the advantages of oil condition monitoring in combination with artificial intelligence (Powrie and Fisher, 1999). This is achieved by monitoring the electrostatic charge carried by the debris in the oil. Even though these systems are in the earliest stages of development, they will provide a significant enhancement by overcoming deficiencies such as the difficulty of detecting non-metallic debris (Tasbaz et al., 1999).

2.2.2 *Gas Path Debris Monitoring*

The overall efficiency of a health monitoring system is based on the accuracy and reliability of the data that it processes. The complexity and the heavy requirements of the new performance-based diagnostic methods have led to research on more practical and definite diagnostic techniques. Therefore, the Gas Path Debris Monitoring diagnostic technique is being presented and is still under development, in order to detect the most obvious indicators of an incipient failure, namely the debris which accumulates in an engine from worn out components.

Since it is not possible to trap any particles, solid or otherwise, behind the jet exhaust, the method, which has many of the same principles as oil debris monitoring, focuses on the detection of the electrostatic charge that this debris carries. Therefore, a baseline/threshold of electrostatic charges for a clean engine is established. In addition, that threshold is deferred for every operational point. Hence, for the given operational condition the system is designed to trigger an alarm when the levels from the threshold are exceeded, which would indicate that a foreign body is passing through the engine (Figure 2.1). The detection of that exceedence is the principal concept behind Gas Path Debris Monitoring (Fisher, 2000; Wen et al., 2011).

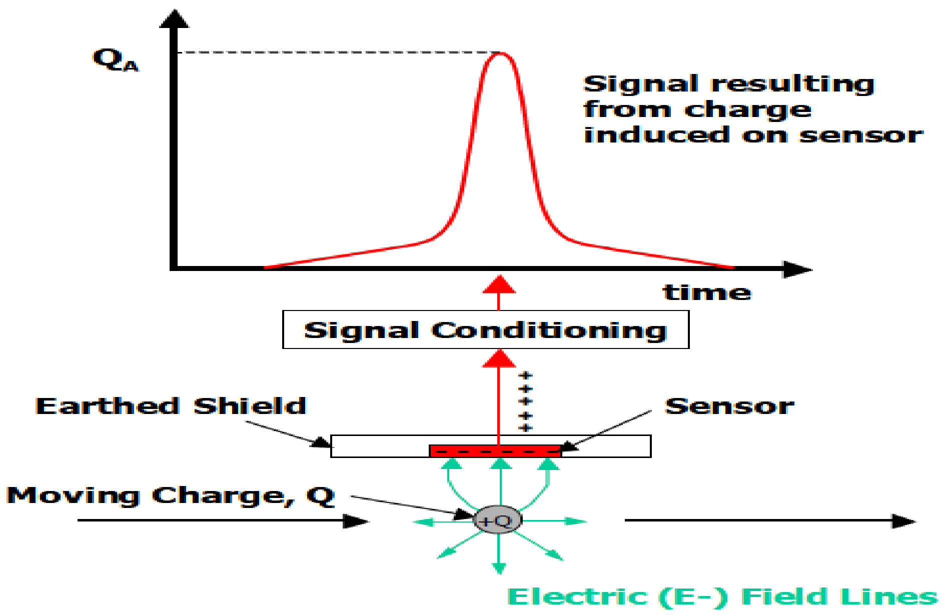

Figure 2-1: Gas Path Debris Monitoring Principle (Powrie and Novis, 2006)

However, not all debris results from the process of wear and it is possible for other debris, such as sand and salt, to enter the engine. Without implying that they are not malicious for the engine's life, it is the responsibility of the system to discriminate between those which are potentially damaging and those which are not. The second category is the ingested objects which could be of many kinds, from solid particles to liquid sprays. This factor requires a significant separation within the complete system architecture, which translates into the existence of two different but co-dependent sensor monitoring units under a generic Gas Path Debris monitoring system. It should be noted that due to the fact that this particular diagnostic technique is relatively new, no detailed descriptions could be found in the relevant publications. For this reason, only the general concepts will be described, as these are implemented in a specific system and as presented in a Gas Path Debris Monitoring System introduced by Stewart Hughes Ltd and undergoing experimental development in the JSF PHM program (Powrie and Novis, 2006).

As mentioned earlier, the separation of produced and ingested debris results in the need for two sensor monitoring units; the Ingested Debris Monitoring System (IDMS) and the Exhaust Distress Monitoring System (EDMS). Because each focuses on a different target group of debris, they are mounted in different places within the jet engine.

The EDMS consists of four main sections: the sensor, the signal conditioner unit, the processor unit and a ground computer. The sensor works as a passive device that detects the exceedence from pre-specified values for electrostatic charges. It is found at the exhaust section in a location with good "visibility", disturbed by as little flow as possible. Such a sensor must be able to withstand those extreme conditions but also allow easy installation and self-maintenance. Recent advances in technology has made the sensors able to combine these functions. Additionally, in order to survive to the conditions in that section of the engine, the stem of the sensor has an integral bellows section.

Whether or not it is the function of the sensor, the charged signals obtained must be converted to voltage signals which can be read by the accepters of the system, which in this case is the processor unit. Therefore there is a dedicated system to provide this function, called the Signal Conditioner Unit or Charge Amplifier, which in many cases is integrated with the sensor module. It also filters the data for the conditioning electronics to ensure their stability whilst monitoring multiple dynamic charges.

The core of the system consists of the airborne processing unit, which is able to read the signals transferred and provides results by assessing and comparing them with the predefined baseline/threshold. This is possible for every given flight condition. An additional asset of the processing unit is the simultaneous correlation of the signals with the operating conditions. This is because some operational conditions may encourage the production of debris, and so promote the occurrence of faults. Hence, by providing information to the operator, the system enables the operator to change the operating scenarios after assessing the data. The particular factors relating to the engine which are monitored in order to assist the correlation are the spool speed and the fuel flow.

In addition, the processing unit is able to interface with other health monitoring units, which focus on different aspects and use different techniques, in order to contribute to the completeness of the overall health monitoring system. A final key point is the ability to store the data in a storage module for further assessment and processing. The latter initiates the ground station computer processing process by utilizing a very powerful computer which is not limited in terms of storage and processing requirements and has greater computational power and so is used for display, trend analysis, report generation and correlation with other engine data. Such computers are used for all the maintenance enhancement systems and appear as an integral part of such systems. All the modules above confirm the OSA-CBM architecture described earlier, and will followed from the maintenance enhancement systems that follow.

Following the same principles, the IDMS has the same general architecture of sensor, data conditioning, data processing and ground processing. However, the goal of the system, which is to monitor the ingested debris, rather than the produced debris, creates differences in the architecture from that of the EDMS. The main difference lies in two sensors mounted in the front of the engine close to the inlet and forming a complete ring around the intake.The sensor construction incorporates three different layers, insulator – conductor – insulator (Powrie and Fisher, 1999).

The following architecture referring to data conditioning and processing has already been described for the EDMS. For the current the processing unit called IDMS Data Acquisition Processing and Storage Unit. The last in order to provide discrimination of the ingested objects it assesses charge and velocity factors. Finally it is the duty of the latter to interface with the EDMU in order to correlate data and provide accurate results (Fisher, 2001).

In addition to object discrimination, the complete Gas Path Debris Monitoring system is able to initially identify the course of the objects via the core or the by-pass by correlating the data (of the two processing units). Different degrees of importance are associated with each, as different possible degrees and types damage could be caused in each condition. Furthermore, the damage caused may or may not be immediate, and certain internal engine deficiencies, such as a surge, could be caused should debris be lodged in the engine. In summary, apart from all the benefits of this method, the primary advantage of this technique is that no special modifications are required in order to integrate the system into existing jet engines.

Despite the fact the EDMS is more mature than the IDMS, the combination of the two provides the operator with the ability to detect faults which can be difficult to spot, such as the degradation of the combustion chamber, by observing the combustion derivatives. This also applies to faults for which other monitoring systems require an observable change at least to one measurable parameter, by which time it may be too late. (Vorilas et al., 1998).

The figures below (2.2, 2.3) show a representation of the system. In the first case the bolt that passes through the engine was detected from both the sensors, which indicates a peak. The cooperation of the two systems suggests that this is not wear-out debris but ingested debris as it occurs in both sensors with a lag, and the verdict is that it is potentially damaging ingestion. Conversely, the figure shows the fluctuations due to salt water ingestion. Even if this is not extremely damaging for the engine, it still causes some changes in the fuel flow and spool speed. Without using the system, the explanation of this change would be difficult (Powrie and Novis, 2006).

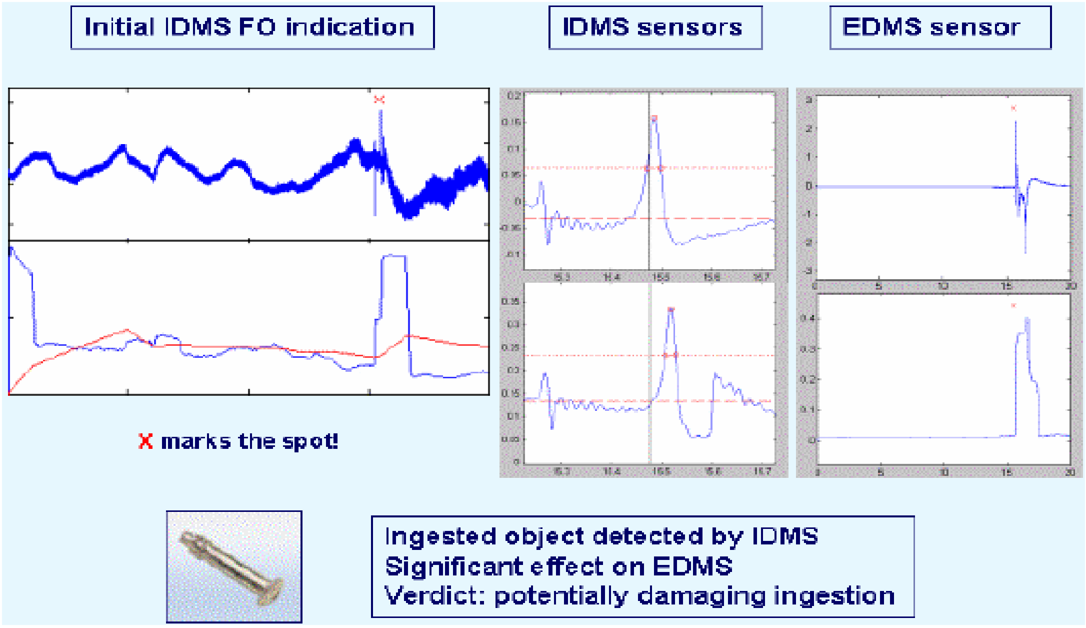

Figure 2-2 Gas Path Monitoring representation (Powrie and Novis, 2006)

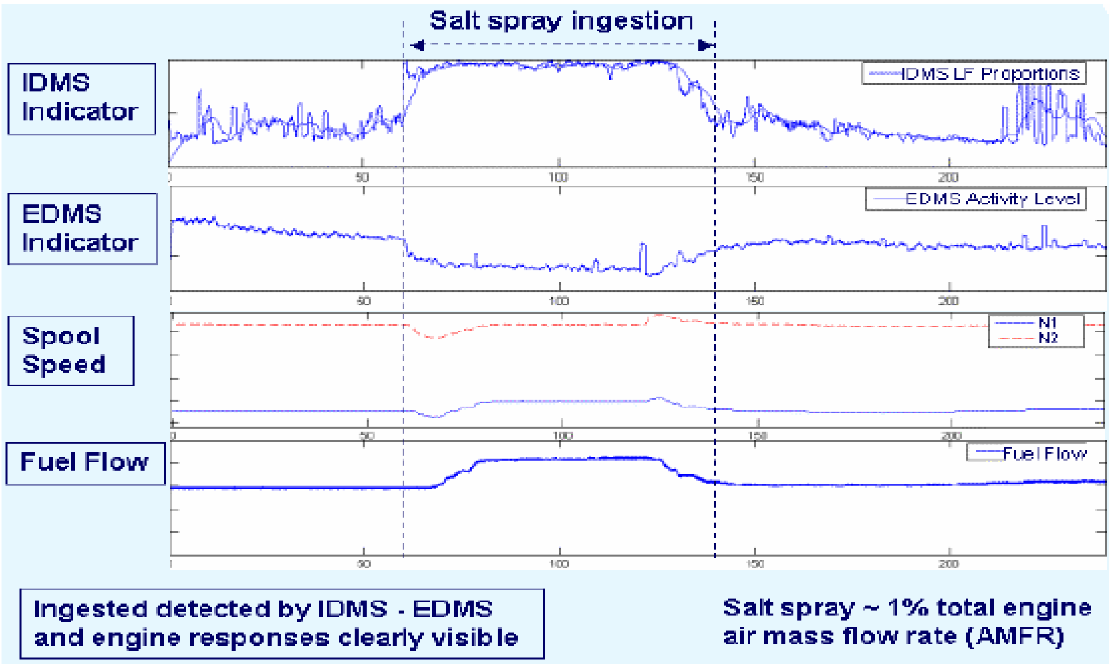

Figure 2-3: Gas Path Monitoring Representation (Powrie and Novis, 2006)

2.2.3 Vibration Monitoring

Vibration is a constant and unavoidable aspect of turbine engines. It can be defined simply as the oscillation or repetitive motion of an object around its equilibrium position; a phenomenon that is exacerbated by the extreme centrifugal and other forces inside the turbine engine. However, it is been proved that a change in the level of vibration could indicate or result in a malfunction of the whole engine or a part of it. Therefore, vibration monitoring is used as a tool to assess the condition and the life of the engine.

Vibration monitoring is classified under the domain of mechanical diagnostic methods. The general concept behind this method does not differ from the majority of modern diagnostic techniques, and so an initial baseline which shows the vibration levels of the clean/new engine is needed. Any deviation from the spectrum of that baseline is an indication of a malfunction, and hence a possible fault. Since its introduction, the technology used for obtaining and processing data has evolved, and several vibration monitoring systems - not necessarily for turbine engines (Pawar and Ganguli, 2007) - have been introduced using different kinds of sensors, processing methods and representation of results (von Flotow et al., 2000; Kapadia and Ray, 1984).

The data obtained come from sensors mounted on and dedicated to specific components. This is because every rotating part inside the turbine engine rotates at a different frequency and so it is difficult for the system to produce accurate results pertaining to the engine as a whole. This is a major deficiency of the vibration monitoring method, and for this reason its installation is limited to some highly critical components or to the principal vibration components from which the vibrations produced are related to malfunctions. Such components are the fan discs, the engine shafts and the shaft bearings. According to Rao (1996) the failures which lead to an increase in vibrations, and which could therefore be diagnosed by vibration monitoring, are the imbalance or misalignment of the shaft; bearing wear, which could be divided into oil film bearings and rolling element bearings, rotor bends, and general faults resulting from stiffness, dissymmetry, component looseness, aerodynamic and hydrodynamic forces, and finally reciprocating forces.

After being generated, the data must be transformed into another form, such as digital signals, and processed by using signal processing techniques. This process is essential in order for the data to be readable and useable for correlation with other diagnostic techniques. The verdict on the state of the condition is usually given in combination with a system that uses a different monitoring/diagnostic method. For example, damage to one of the bearings will result in an increase in the amplitude of vibrational components, and a possible increase in the temperature measured on the casing around that bearing. However, because the aim of this research is simply to provide a general guideline to the method, further information relating to the obtaining, processing and assessing of vibration monitoring data, and for the vibration monitoring science in general can be found in Randall (2011).

2.2.4 Borescope Inspections

Despite the evolution in engine diagnostics methods and systems, any conclusion arrived at by means of other methods should be combined with visual indication of faults whenever possible. This could mean that whichever is the verdict from the maintenance enhancement systems like those that will be described in the following chapter, the last decision should be taken after a visual confirmation. Visual condition monitoring techniques have been developed to provide a final and definite result, and have been in use since the inception of the turbine engine. With the exception of simple checks, visual inspections/diagnosis for aircraft engines can be taken to mean borescope inspections.

This is perhaps the simplest technique for engine diagnostics, and it is the tools used in order to perform a borescope inspection that have evolved since its introduction, rather than the general concept. The fact that it entails the use of special equipment classifies this technique as a diagnostic method and separates it from the routine visual checks. Even if the tools used are state-of-the-art, the results obtained depend largely on the skill of the operator performing the inspection.

The general concept of the method is very simple to describe, but in fact is very difficult to accomplish, and so needs years of experience. Initially, the borescope inspection refers to every type of engine, be it piston or jet, and provides visual access to covered or hard-to-access parts of the engine, without the need to remove any of the components, because the engines are equipped with special ports for that purpose. A typical port configuration and areas of a jet engine accessible via a borescope are shown in Figure 2.4 below. It is possible for the mechanic to check the engine for failures which may impact upon the safety of the next flight, such as gas or fuel leaks, and also the security of the pipes and the condition of the coating.

However, as a sophisticated diagnostic tool, the borescope inspection has the ability not only to validate results from other diagnostic reports, but also to detect faults that are “undetectable” by other performance or mechanical condition monitoring techniques. This is because very sophisticated diagnostic methods have not yet been fully implemented on the maintenance enhancement systems, and are in still at the experimental stage. Hence the human visual check is still necessary. Examples of such faults are the presence of nicks or the beginnings of cracks, or erosion and corrosion. This type of fault needs to be present in large amounts in order to be detected; something that can be “corrected” by performing a borescope inspection.

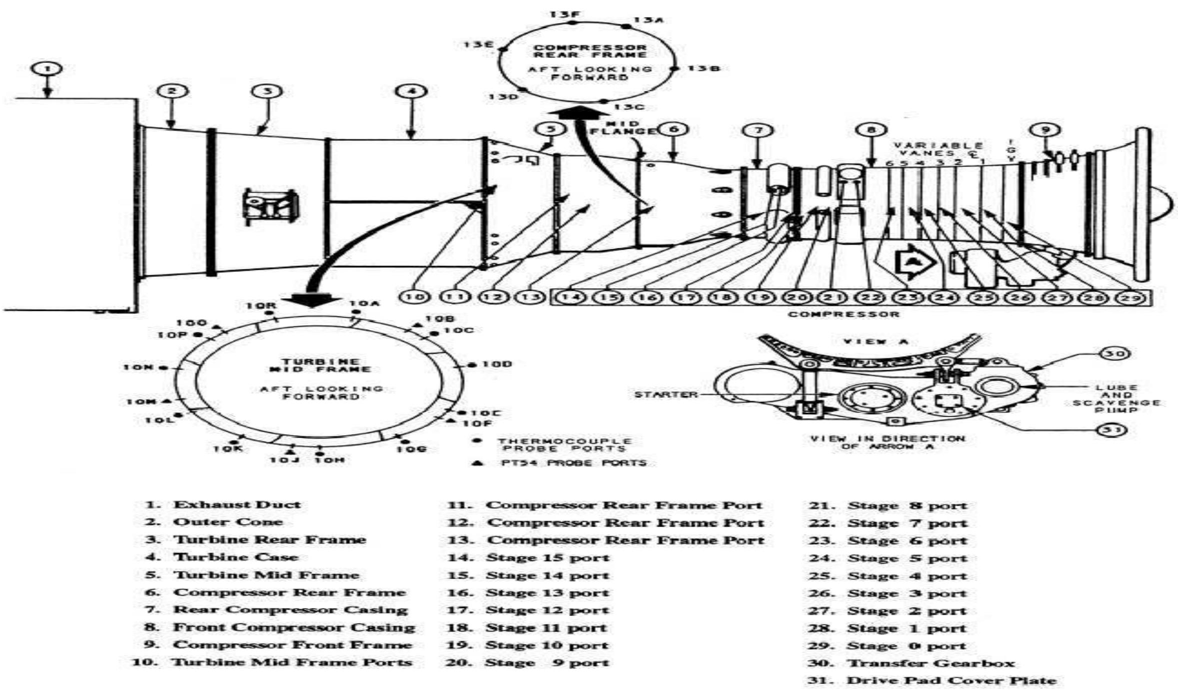

Figure 2-4: Typical Engine Port Configuration for Borescope Inspection (Lufthansa Technical Training, 2007)

The equipment that is used for a borescope inspection is of two kinds; rigid and flexible endoscopes. These are generally simple devices that transfer the image from the spot and the location of the fault to the human eye. The most obvious difference between them is the degree of flexibility that these instruments offer. The flexible endoscope is preferable because it can reach parts of the engine which are more difficult to access (Figure 2.5).

Figure2-5: Typical Flexible Endoscope (www.lasaero.com)

2.2.5 Engine Usage Monitoring Diagnostics

In addition to all the other condition monitoring and diagnostic methods which can be applied to aero engines, there is another general category of techniques for assessing the remaining life of the engine which does not share the complex processing and the real time constant data acquisition of the others. This category relates to engine usage diagnostics; methods that are incorporated at the engine's design stage in order to create an initial maintenance planning guideline proposed by the manufacturers, but which are also employed during the service of the engine in order for the operator to adapt the maintenance planning to the engine's current usage conditions. By employing engine usage monitoring methods, it is not possible to predict unexpected incipient failures, or the early indications of such failures. However, because they predict the life of the component based on the experimental data of materials' properties and loads, they can be classified as mechanical diagnostic methods.

In order to completely understand engine usage monitoring diagnostics, it is essential to first identify the most important and unavoidable mechanisms of failures for an aero engine, namely creep and fatigue. Briefly, creep is the deformation of the components that are exposed to extreme thermal and stress forces, and is time dependent. That failure cause will be later the topic of research in the following case study. Fatigue is the deformation to which the materials are subjected due to extreme stresses, and can be divided into High Cycle Fatigue (HCF), Low Cycle Fatigue (LCF) and Thermal Fatigue (TF).

Knowing that, such failure modes will definitely occur at least once during the service life of the engine, as the engine's components become degraded, and methods for calculating the life of each component have been developed. These calculations require complete data pertaining to the component's geometries and the properties of the materials of which each component is made. In order to complete the calculation, the condition parameters at which the engine is operating at each moment, such as spool speed, turbine entry temperatures, fuel flow etc., are also required.

After having collected all the data required, the calculation can begin, and can provide results relating to the remaining life of the components for the given and optimum usage conditions. However, this is the major disadvantage of the method because the given conditions for calculation are assumed, and differ significantly from the actual conditions. An example of this is shown in Figure 2.6. It is clear that there is a considerable difference between the given performance data (continuous line) and the actual data (dashed line).

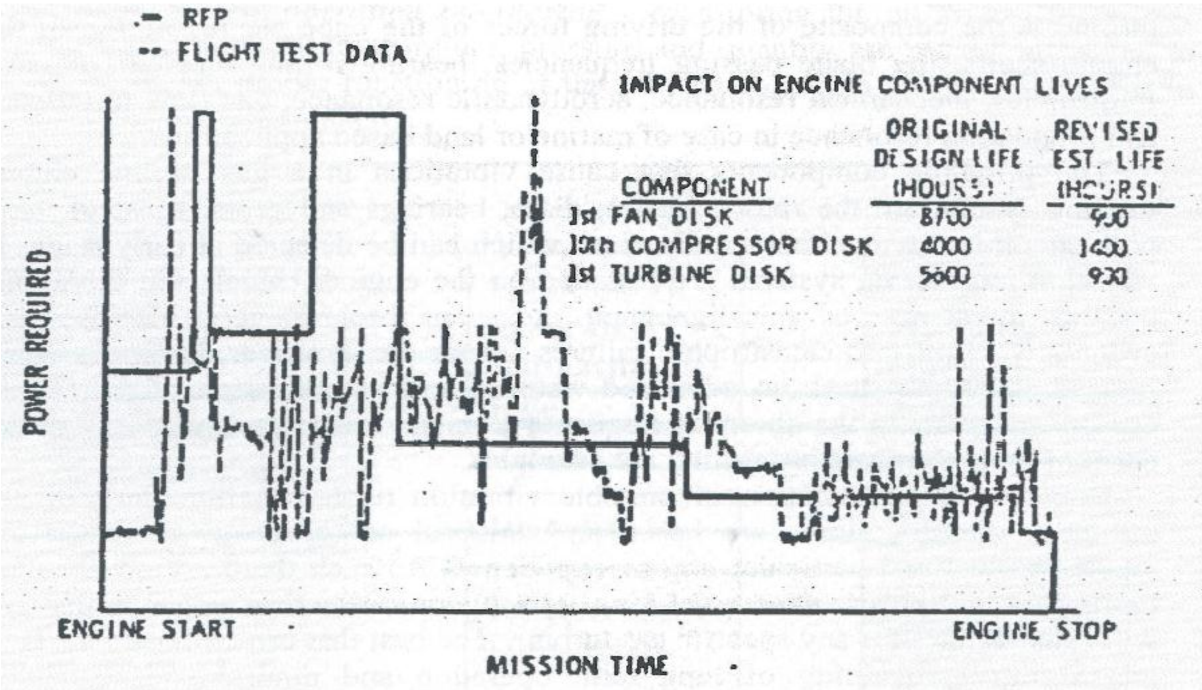

Figure 2-6: Assumed Thrust (continues line) and Real Thrust (dashed line) (Li, 2008)

Moreover, for an accurate calculation of the remaining life of the component, the atmospheric and environmental conditions at which the aircraft is operating are of great importance, and hence a single calculation could not be applied to all the engines, even of the same model, because they could be located in different places, with different environmental conditions, and moreover could be subject to different loads.

For example, the turbine blades will be affected if the atmosphere has a low density of air because in order to provide the same amount of desirable thrust, the engine has to increase the fuel flow, and thus also the turbine entry temperature. The latter has a direct effect on the creep life of the components after the combustion chamber in terms of load and taking as an example an aircraft operating in a less busy airport with long runways versus an aircraft operating in busy airports without the convenience of a long runway.

A condition monitoring system that utilises the benefits of engine usage diagnostics combined with other diagnostic methods will result in a significant improvement to the engine health management because the operator could confidently determine the real remaining life of the engine based on its usage and material properties, but could also exploit the result from the other diagnostic methods in order to cover the unexpected failures that are not related to material properties. Later in this research a case study will be presented, concerning the effects of reduced thrust in engines health, by utilizing the engine usage diagnostics and especially the creep life analysis.

2.3 Performance Diagnostics

2.3.1 Gas Path Analysis

The narrow monitoring spectrum of the mechanical diagnostics and the disadvantage of focussing on input data from specific components rather than from the whole engine, led to the development and establishment of performance diagnostics, with their initial method being Gas Path Analysis. It was first introduced in 1967 and apart from the mechanical diagnostics, was able to improve the inaccuracies and deficiencies that resulted from previous non-mechanical diagnostic methods, which were based on the analytical model and were only able to detect a single fault. The most popular of these were the Fault Tree and Fault Matrix methods (Li, 2008).

In contrast to these methods, the greatest advantage of the Gas Path Analysis method is the ability to detect more than one problem simultaneously and to provide quantitative results. It was developed by Louis A. Urban, who describes it as: "a mathematical technique which estimates overall engine performance and individual module and sensor performance from any specific set of engine measurable parameters, such as temperatures, pressures, rotor speeds, fuel flow etc, through the aero-thermodynamical relationships which exist between them" (Urban, 1972).

Before proceeding to a description of the method, it is important to understand what constitute the measurable and non-measurable parameters. The measurable parameters are those for which the acquisition of their values is possible via measurements with instruments. The range of those parameters may differ from engine to engine depending on the technology implementation on it, however, typical available parameters are shaft speed, fuel flow, total and static pressure, total and static temperature, the outcome of the previous two, namely the total inlet airflow, the shaft torque and of course the power settings. Very importantly, the flight condition parameters are also in this category. For the case of Gas Path Analysis, these are called dependent parameters, and they refer to specific components or sections condition/behaviour, rather than to the whole engine (Provost et al., 1994).

Conversely, the parameters that refer to the whole engine, which cannot be directly instrumented and must be calculated, are the independent parameters, or performance parameters, such as the flow capacity, isentropic efficiency etc. An example of the difference between them is the following. It is achievable to measure the pressure or the temperature of a section, but it is not possible to take a direct instrumentation of its efficiency, which is the ratio between theoretical and actual temperature for a given pressure ratio (Vorilas et al., 1998).

The problems encountered by gas path wetted components are well known, and centre around erosion, corrosion, fouling, flight loads, foreign and domestic object damage, rubbing wear, and finally thermal distortion (McEwing et al., 2002). It is been observed that the appearance of one or more of these has an effect on the engine's calculated performance parameters, noted earlier as independent parameters. An example of this is a change in isentropic efficiency, and/or air pumping capacity may result from compressor faults that experience one of these problems. Because the independent parameters are outcomes of the dependent parameters, there will be changes in these as well, with the only - but very important – difference being that we can measure them. Deviation from a threshold measurement of the dependent parameters will give us the suspicion of deterioration, or else a fault signature.

The fact that the independent parameters cannot be directly measured makes the correlation with the dependent parameters essential in order to obtain the desired results. It is in the relationship between those two categories that the fundamentals of Gas Path Analysis lie. The more analytical process begins with the creation of the engine's analytical performance model. This is accomplished by matching individual component characteristics and aero-thermal relationships and it is expressed in terms of dependent and independent parameters. The individual components are the turbine, the compressor etc and the measured values from them and the aero-thermal relationships are well known laws of conservation of energy and mass (Ab Halim et al., 2005).

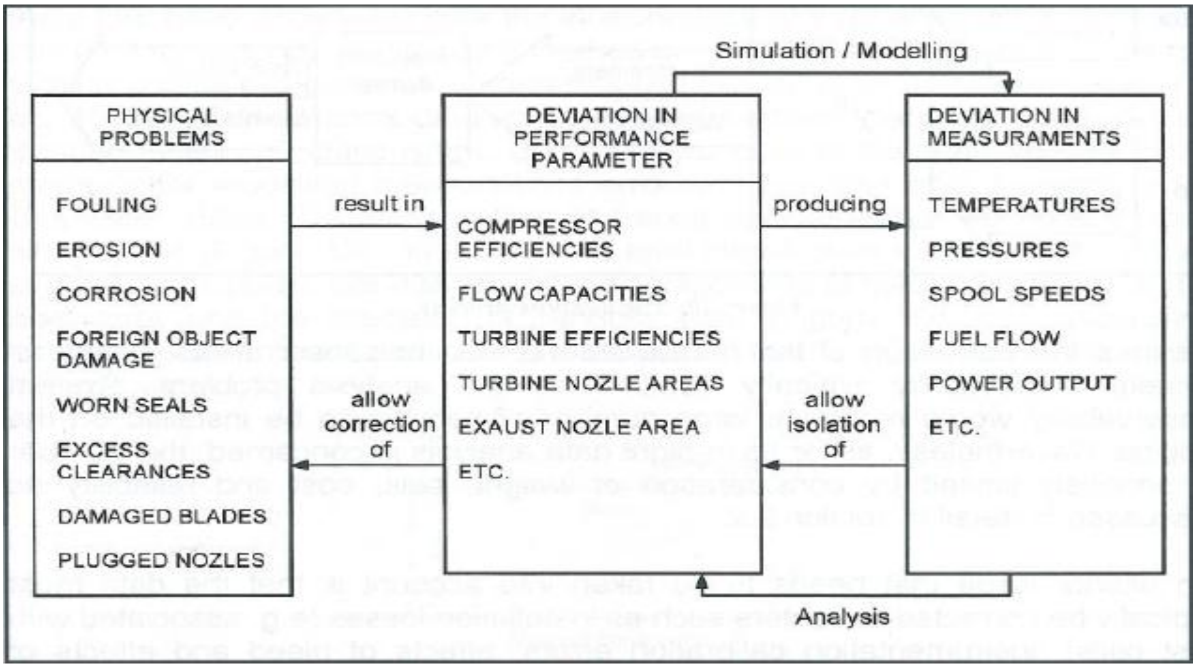

Figure 2-7: Overall Concept of Gas Path Analysis (Ab Halim et al., 2005)

It should be noted that Gas Path Diagnostics assesses changes in performance and not in the absolute values, which means that the result comes from the comparison of the observed and the nominal value.

2.3.2 Linear Gas Path Analysis

The concept described above led to the creation of the simplest form of Performance-Gas Path Diagnostics, which is Linear Gas Path Analysis. According to this method, the linearity between the dependent and independent parameters is assumed even though this is inaccurate, and constitutes one of the greatest disadvantages of the method. However the method begins with the basic equation 2.1:

$$\vec{Z} = h * (\vec{x}) \qquad (2.1)$$

Where z is a vector of M measurements ($z \in R^M$), x is the vector of N performance parameters ($x \in R^N$) and h is a vector-valued function provided by the performance simulation model. In the same way the deviations of the parameters are also expressed, thus giving the following equation, which is also known as linear Influence Coefficient Matrix (ICM) or Exchange Rate Table. It should be noted that in order to create the ICM we must assume that the change in the Independent Parameters is relatively small. Additionally, for a given steady-state operating point we can linearize the equation using Taylor series, which will by then expressed expanded as matrices (equation 2.2).

$$\Delta\vec{z} = h * \Delta\vec{\chi} \qquad (2.2)$$

If we invert the previous equation to find the presence of any fault then we create the Fault Coefficient Matrix (FCM) or diagnostic matrix, which is expressed as equation 2.3, and for one more time we assume that the matrix is invertible. For that inversion the Taylor series is used, in which the Higher Order Terms are neglected. However, even with the previous assumption the inversion is possible only if the number of component parameters is equal to or smaller than the number of measurements (N≤M). In fact it is very difficult to achieve that equalization and so estimation methods are applied, which are the evolution of the initial Linear Gas Path Analysis (Provost et al., 1994). Such estimation methods are the Kalman Filters and the Weighted Least Squares, applied for linear and non-linear Gas Path Analysis.

$$\Delta\vec{\chi} = h^{-1} * \Delta\vec{z} \qquad (2.3)$$

It should be noted that the name Gas Path Analysis refers to the methods based on the initial method/idea provided here, unlike Gas Path Diagnostics, which refers to different methods that use gas path performance models. Because Gas Path Analysis is a very representative and very commonly used example, it is described in this research, although it is felt that there is no benefit in providing a detailed description of all of them (Ntantis et al., 2008).

The obstacles that Gas Path Analysis faces in order to provide accurate results can be categorized into five types. The first is Measurement Noise, which is by no means a negligible parameter. Because in some cases the level of noise could be equal to the component's measurement variations, special attention must be paid. The second type, the Sensor Fault, works as a supplement to the first because in some ways the avoidance of it compounds the inaccurate results from the first. Typical problems that occur with the sensor fault problem are bias and sensor drifting, inaccuracies resulting from calibration errors, and the effect of bleed and Reynolds numbers (Marinai et al., 2004).

However, apart from the actual problems resulting from sensors, a third obstacle arises within the same category, and refers to the choice of measurements. In practice this means that the detection of every fault needs a special instrumentation and sensor set in order to be diagnosed. However, in most cases the Gas Path Diagnostics programs must work with the instrumentation which already exists on-board; something that has not been selected to fulfil Gas Path Analysis accuracy requirements. The measured parameters of an aircraft engine are listed in the following chapter.

In addition, the fourth obstacle is the number of measurements, which has to be equal to the number of components parameter. This is because in order for that to be achieved, sensors have to be installed in every component, which is difficult due to weight, bulk, and most importantly cost for the sensor implementation and sensor maintenance; factors that could reduce the possible benefit from the monitoring method because of their costs. However, installing a large number of sensors is not always beneficial as there is also a greater probability of sensor fault, and hence reduced reliability of the overall monitoring system (Marinai et al., 2004). The obstacles referring to the sensors and to their deficiencies are known as observability problems, and an analysis of these must be undertaken prior to attempting any diagnostics. Further information can be found in Gelb (1999).

In any case, special attention should also be given to the baseline data, which are the initial data from which any deviation will trigger the mechanism. Unfortunately every single engine, even of the same type, seems to display different behaviour, although always at acceptable levels, and as such it is considered difficult to customize a general data baseline for a given type of engine. Therefore it is essential to conduct a test run of every different engine to establish its own unique set of baseline data (Li, 2008).

Finally, and maybe most importantly, the relation between measurements and component parameters is everything but linear The final point is that linear Gas Path Analysis is no longer useful when the deteriorations are too great and cause the engine to operate (different operating point) in a way that is different from that for which the matrix was pre-calculated, which is when non-linear Gas Path Analysis comes into play.

However, despite the deficiencies described here, linear Gas Path Analysis can be a very helpful initial tool for fault isolation and quantification estimations. This is supported by the fact that the overall procedures (Figure 2.8) are simple, with rapid results. No special complexity of maths is required, and no iterative calculations are needed.

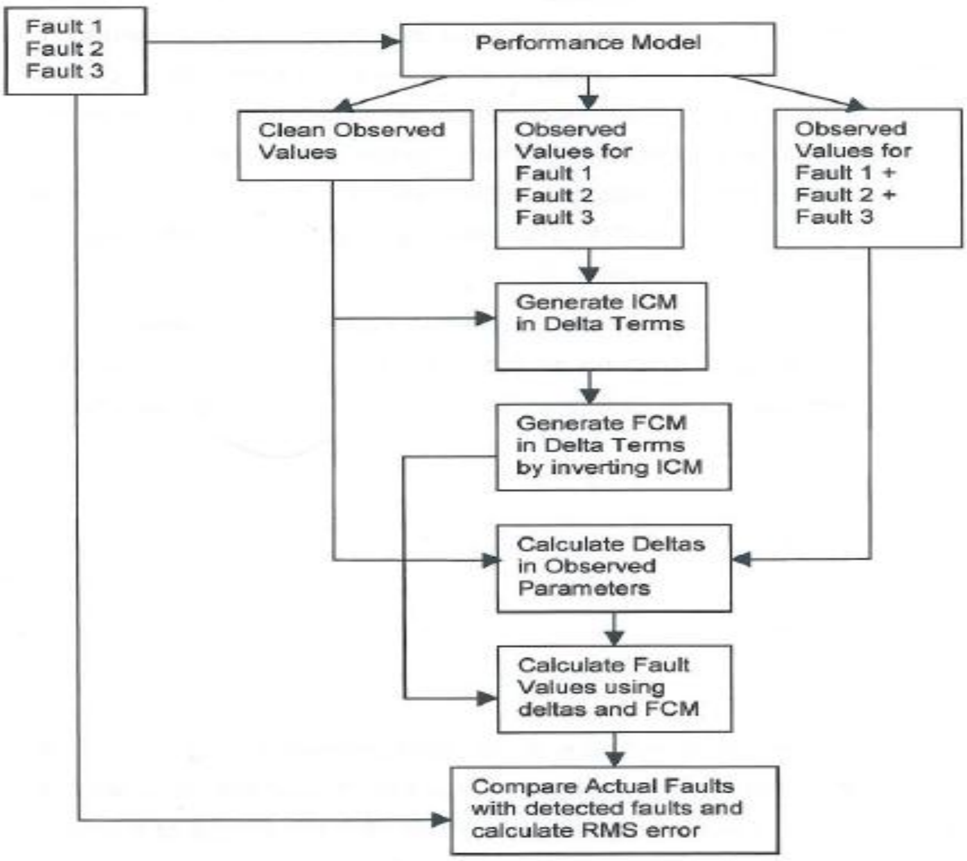

Figure 2-8: Schematic Representation of Gas Path Analysis Procedures (Source: Author)

2.3.3 Non Linear Gas Path Analysis

As was mentioned earlier, the relationship between dependent and independent parameters is not linear. Hence, by assuming linearity in those on the one hand we will take some results of the existence of a failure, but the inaccuracy that these results will display may be of the same magnitude as that of the actual fault. One solution, and a more accurate approach to that problem is arrived at by solving the equation described previously by using an iterative mathematical method. In the literature the most frequently used iterative method was found to be the Newton-Raphson method. The actual benefit of the iterative method is that it takes into account the Higher Order Terms (HOT), and hence the range of fault comes with greater accuracy.

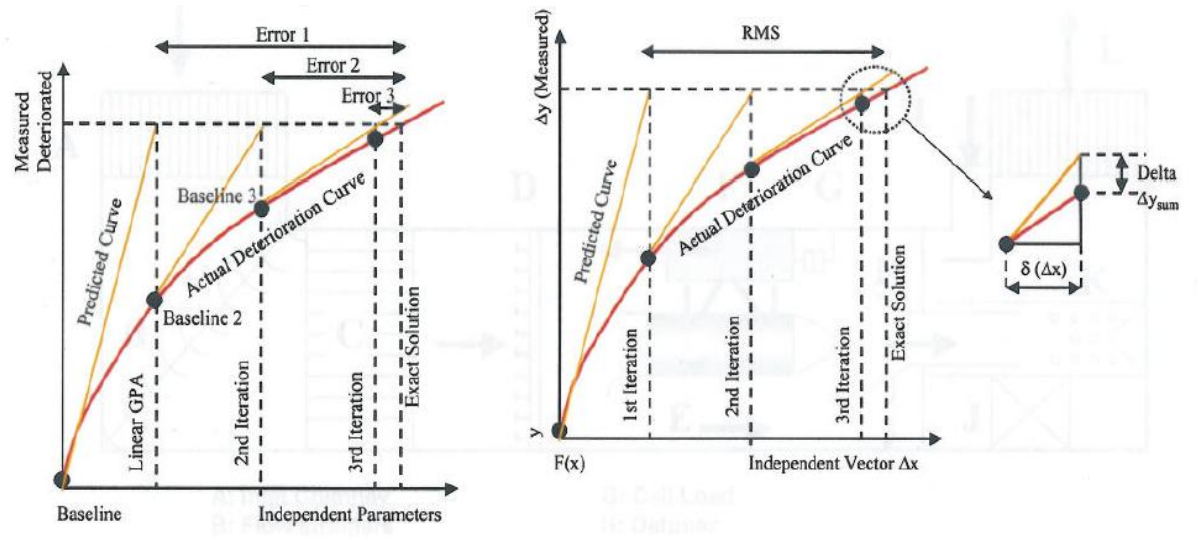

Figure 2-9: Non linear Gas Path Analysis Concept Representation (Garcia Perez et al., 2003)

The overall concept is easy to describe by explaining the graph shown in Figure 2.9. Fundamentally, the Gas Path Analysis diagnostic method is based on the principle that the fault is defined by the deviation of independent and dependent parameters. The top line inside the graph shows the deteriorated performance, the x axis shows independent parameters, and the initial baseline shows calculated performance. The change from the baseline performance to the deteriorated performance line is the delta dependent parameter column matrix, which multiplied with the initial FMC will give us a calculated delta for the independent parameter line, which is shown as point 2.

66% of the independent parameter change is taken to form point 3. This is called Damping factor λ and the default is 66%, but the user can peak values from 0 to 1. The selection of that parameter is very important due to the fact that the smaller the λ, the more vulnerable is the system to get an oscillating loop around the solution (Vorilas et al., 1998). Continuing the previous process adversely, we are trying to create a new base line (interim 1) by calculating the changes to the dependent parameters, as now we know the delta of the independent. This process is continued until the convergence criteria pre-specified by the user are satisfied, which will determine the desired difference between the calculated and measured parameters (Garcia Perez et al., 2003). It is very clear that by setting a new performance baseline every time the range is shortened will also increase the accuracy of the method. This is evident in the graph simply by looking at the difference between 2 to 1, which is the initial condition (linear GPA) and 8 to 1, which is the final outcome from the non-linear GPA (Vorilas et al., 1998).

Here, as well as almost everywhere, some very important assumptions must be taken prior to attempting any Gas Path Analysis. The first is that we consider the flow as one-dimensional and steady-state, meaning that the performance parameters remain constant at every point of the gas flow. Additionally, the flow is considered as adiabatic and free of cooling losses. Finally, some level of noise free operation due to ideal instrumentation is considered (Garcia Perez et al., 2003).

The deficiencies that an integrated diagnostic system may have if it uses only one diagnostic method is confirmed in the case of Gas Path Analysis. This is because the method is unable to provide any results for potential specific failures. Some of these are failures due to oil system leaks, bearings failures, creep in turbine blades and fatigue failures in mechanical elements in general, whose impact on the overall engine performance, because of their primary stage, could be negligible. However for all of the above, there is a threshold under which a change in Gas Path Analysis detected parameters will be detected. For example, when the bearing failure is significant, it will result in excessive spool friction, but by that time it may be too late (Ab Halim et al., 2005; Smith, 1996).

2.4 Artificial Neural Networks

It is commonly accepted that the most powerful processor in existence is the human brain. Even if we omit or degrade the ability of the brain to process data very rapidly, in a way that no modern processor can, the human brain has the unique and very important ability to perform certain functions effectively by using examples from stored knowledge. This could be defined as experience, and it is the most desirable function that a diagnostic tool could perform.

As such, there have been considerable efforts on the part of scientists to create systems that emulate the human brain by means of data processing, storage and training. Such systems are generally termed as Artificial Intelligence Systems. Concerning applications related to turbine engine diagnostics, maybe the most representative example of such is Artificial Neural Networks, which appear to be very effective, with promising future applications. Artificial Neural Networks are classified as performance diagnostic methods, but they have one significant difference from other methods, which also constitutes their advantage, namely, the ability of such a system to effectively train itself, without the need for any pre-defined rules or models (Li, 2002).

In order to describe in a simple way how an Artificial Neural Network diagnostic system works, it is necessary to identify the most important elements of such a system. At the centre of this kind of diagnostic system is the neuron; in other words, the data processing unit, each of which is assigned to a different function. Each has the ability to process the data in parallel and produce an output of a weighted sum. The different weights come from the different algorithms that every neuron must have in order to perform its function. For diagnostic systems that use Artificial Neural Networks as the diagnostic method, numerous neurons are needed.

Because of the requirement for many neurons, the system is divided into layers, each of which has its own dedicated neurons. Connections, or synapses, between the neurons enable the flow and processing of information, depending on the role of the particular neuron. These synapses have their own weights, which are derived from the different algorithms pertaining to each connection. Essentially these weights provide the neurons with the most appropriate data for processing and storing information (Byington et al., 2002). Additionally, and very importantly, depending on those connections there is a distinction between the types of network. This distinction concerns the type of the data flow, which could either be a forward flow to every successive layer, or both forward and backward flow, to enable the system to be trained via the feedback process (Jain et al., 1996).

The presence of these layers is essential and a general description, which is the purpose of this thesis, is sufficient. Put simply, there is an input layer, an output layer, and between them are one or more hidden layers. As mentioned above, several neurons are located within every layer. The need for fault detection, isolation and quantification creates the structure of these layers, and particularly the input and the output layer. Hence a complete network is constructed from several sub-networks which are dedicated to specific components, and also to specific failure modes of every component.

In order for a system to be operational it must pass through two general phases. The first phase is the training mode. In this stage the network is fed not only with the kind of the data that it will process during its life, but also with the sort of failures that might occur. Generally the sets of data required for the training phase are the training data, the target outputs and the test data. After that phase is completed, the system is in the recall mode, in which it is able to perform the diagnostic/prognostic requirements. These two general phases are part of every kind of network (Joly et al., 2004).

Even if there are distinctions that separate the artificial networks into many categories, it is the authors' opinion that the basic distinction concerns the function of the network. As was previously noted, the three requirements of fault detection, isolation and quantification provide the shape of the basic structure of the network; hence there are Artificial Neural Networks for each of those functions, with the difference lying in the input and output layers, or more specifically, the neurons that are within those layers (Patel et al., 1996b).

Beginning with fault detection the network architecture has a simple form, consisting of three layers, with the last output layer having only one neuron. In general the purpose of such a network is simply to indicate whether there is degradation, which might indicate a fault for the engine. The input layer has the same number of neurons as the monitored components or parameters. Conversely, the output layer has only one neuron, which shows whether or not the engine has a fault, with discrete signals of zero (0) and one (1) for example (Figure 2.10). However for all kinds of networks the input layer should have an equal number of neurons as the required monitored parameters (Lu et al., 2001; Simani and Fantuzzi, 2000).

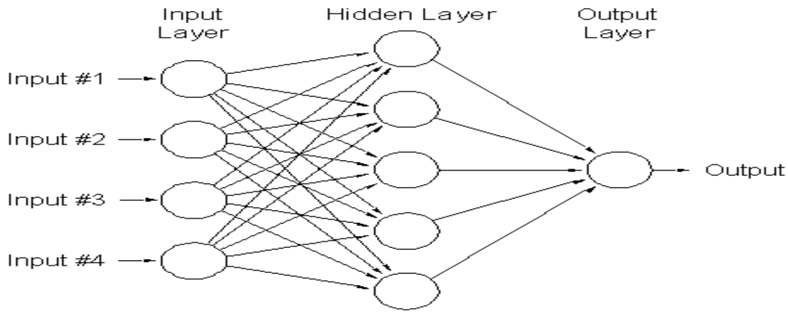

Figure 2-10: Fault Detection network (Li, 2008)

The same three layer architecture is applied for the fault isolation, the only difference being that here the output layer has an equal number of neurons as the components or parameters under investigation, and separate results are needed for each of them. The discrete signal of 0 and 1 can also apply here, but in this context it indicates the specific components which are faulty, and not the sum of faults for the whole engine. The difference between this network and the fault quantification network is that the output signals will be more complex from a mathematical point of view than the discrete signals of 0 and 1 (Li, 2008).

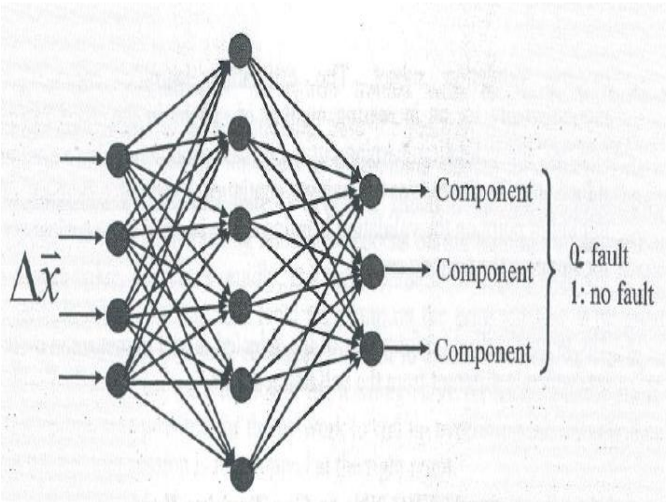

Figure 2-11 Fault Isolation Network (Li, 2008)

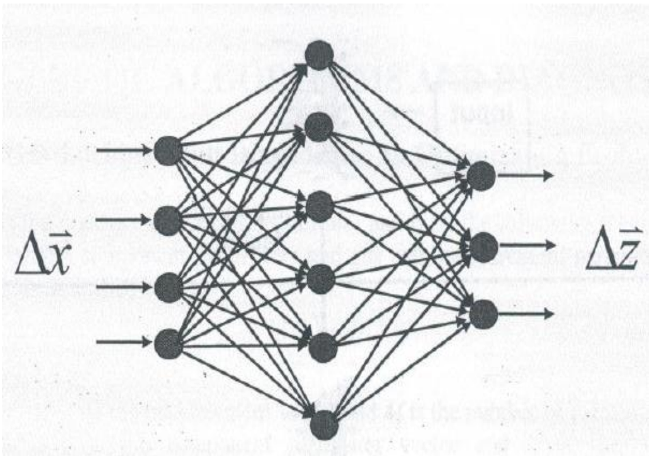

Figure 2-12: Fault Quantification Network (Li, 2008)

The rest of the separations concern factors like the kind of training mode, the type of data flow etc. As far as the Artificial Neural Networks that are used for gas turbine diagnostics are concerned, the most common types in use are:

- Feedforward back propagation neural networks
- Probabilistic neural networks
- Self-organising maps
- Learning vector quantization networks
- Counter propagation networks
- Adaptive resonance theory networks
- Resource allocating networks
- Recurrent cascade correlation neural networks

The first category of feedforward back propagation neural networks has been found in many publications to be the best solution. It should be mentioned that the analysis of each of the above is beyond the scope of this research, and that further information about each type can be found in the references given in Li (2002).

In summary, whatever the kind of network, Artificial Neural Networks have the advantage of being able to perform parallel processing of data and operating satisfactorily, even with limited information. In addition, they are very tolerant of noise and bias problems, and most importantly for a turbine engine, they can handle the non-linear relationships of dependent and independent parameters. On the other hand, the disadvantages are also important. The main disadvantage is that the optimal network architecture/structure for a given problem is not known from the optimum data sets for monitoring. Finally, the difficulty of defining the optimum training criteria is something that limits the system (Joly et al., 2004; Ogaji and Singh, 2003; Patel et al., 1996a). A typical diagnostic system based on artificial neural networks is shown in the following Figure 2.13.

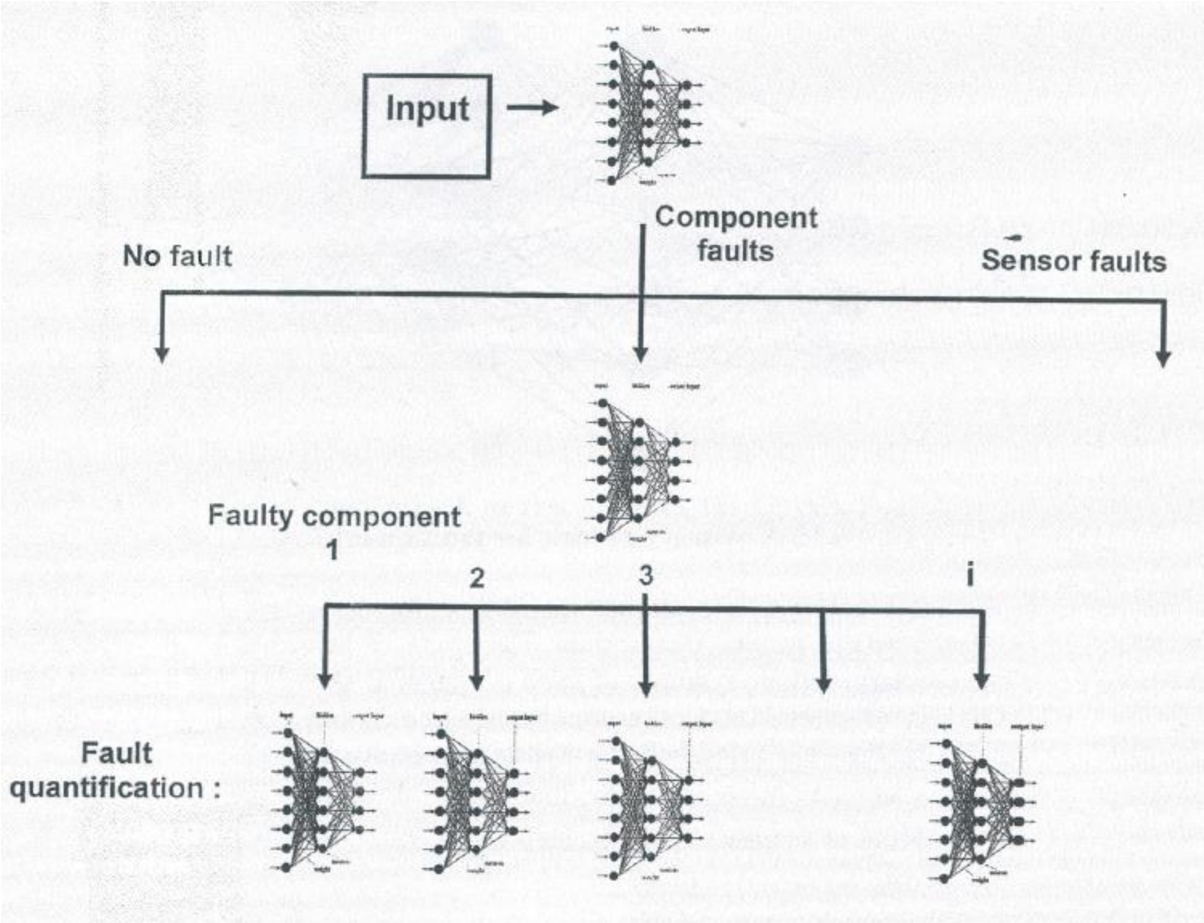

Figure 2-13: Artificial Neural Networks Diagnostic System (Li, 2008)

Chapter 3: Maintenance Enhancement Systems

3.1 Introduction

In a world which demands cost reduction and optimum resource exploitation in any kind of industry, the state of the art technology of condition monitoring is becoming a necessity. Therefore, modern aircraft have no option but to comply with the requirements of accurate and on-demand maintenance, along with optimum resource organization. That concept is totally supported by the development, from the aviation industry, of systems that are hosted on the aircraft and dedicated to maintenance optimization.

Therefore, the creation of maintenance enhancement systems was established with the two major representative examples of the Central Maintenance System (CMS) and the Aircraft Condition Monitoring System (ACMS). The general goal of maintenance optimization is shared by both systems. Nevertheless, apart from the different processes that each system uses, the paramount difference lies in the time window to which each of them refers. Briefly and in general terms, the CMS is dedicated to instantly reporting faulty systems for line maintenance, while the ACMS performs trend monitoring in systems/components whose maintenance depends largely on usage and time. Both systems consist of an actual, existing application of the OSA-CBM architecture. Moreover, the diagnostic/prognostic methods that were described previously are implemented here.

The initial monitoring platform that was necessary in order for the systems to be functional was already in existence due to the fact that system monitoring was already taking place in terms of the control and airworthiness of the particular system. Hence, the newly developed systems should be able to utilize the monitoring capabilities and the data obtained from those systems, and use them after in order to provide output results which would concern only the maintenance of those, by providing fault reports with additional links to the maintenance documentation. This chapter is dedicated to the detailed description of those systems and their functions as they are found hosted in Airbus aircraft, although similar systems exist for other manufacturers (Nicholson and Whitfield, 1990; Aslin and Cole, 1988; Gorinevs.ky et al.).

3.2 CENTRAL MAINTENANCE SYSTEM

3.2.1 Purpose of the System

The modern version of reactive maintenance comes with the utilization of the Central Maintenance System (CMS), which can be found on several modern aircraft, and is the first of the two maintenance enhancement systems. The basic purpose of the system is to provide the maintenance crew with information that comes from faulty systems on the aircraft regarding their condition by centralizing and memorizing all the data necessary for maintenance, augmented by fault reports from the crew.

The generated data that indicate faults within every system come from internal system processing provided by the so-called BITE. The Built in Test Equipment (BITE) is a “device” installed in almost every individual system on the aircraft in order to detect, isolate and memorize the faults that occur in that specific system. Depending on their criticality, the individual systems of the aircraft are divided to Class 1, 2, 3 systems. That distinction is also followed in the BITE of every system, which is connected to the Central Maintenance Computer (CMC) and determines the available report options that will be shared with the CMS. The responsibilities of a BITE system are:

- To detect faults affecting the system
- To identify the fault at the level of the LRU
- To make the distinction between faults at system level (internal faults) and faults at aircraft interface level (external faults)
- To memorize the necessary maintenance information
- To provide dialog with the CMS for the test functions (Airbus Industry, 2010d)

All the documentation necessary for fault identification, along with the associated maintenance, is provided by the system’s reports. It is capable of operation in flight and on the ground, and just by choosing different menus (Figures 3.1, 3.2), the CMS operator can create system status reports, which are either printable or transmissible, depending on the occasion. Its use is usually limited to line maintenance, however, using it at a higher level allows for more detailed maintenance in the base-hangar maintenance (Airbus Industry, 2010c).

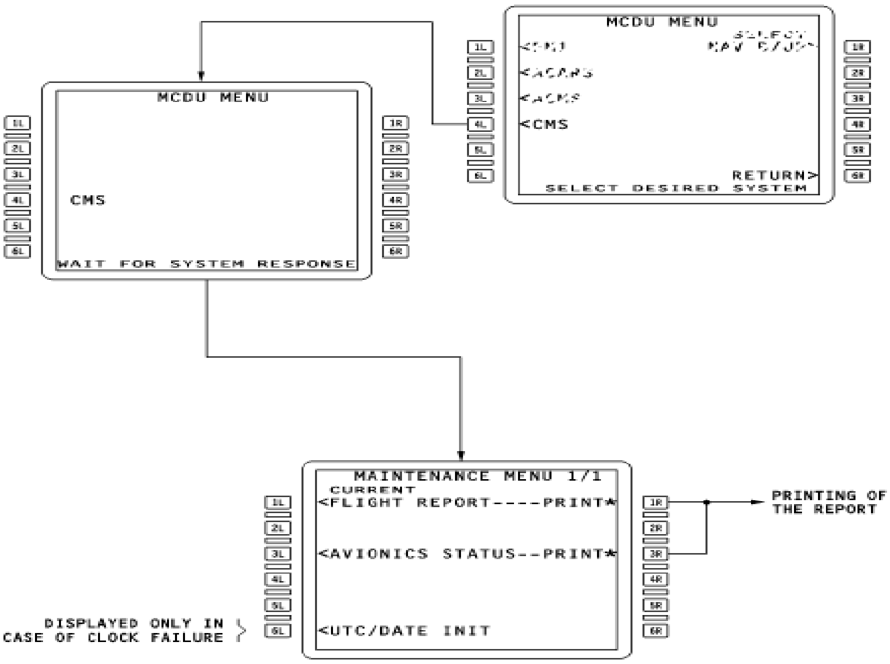

Figure 3-1: In-flight report generation menu (Airbus Industry, 2010c)

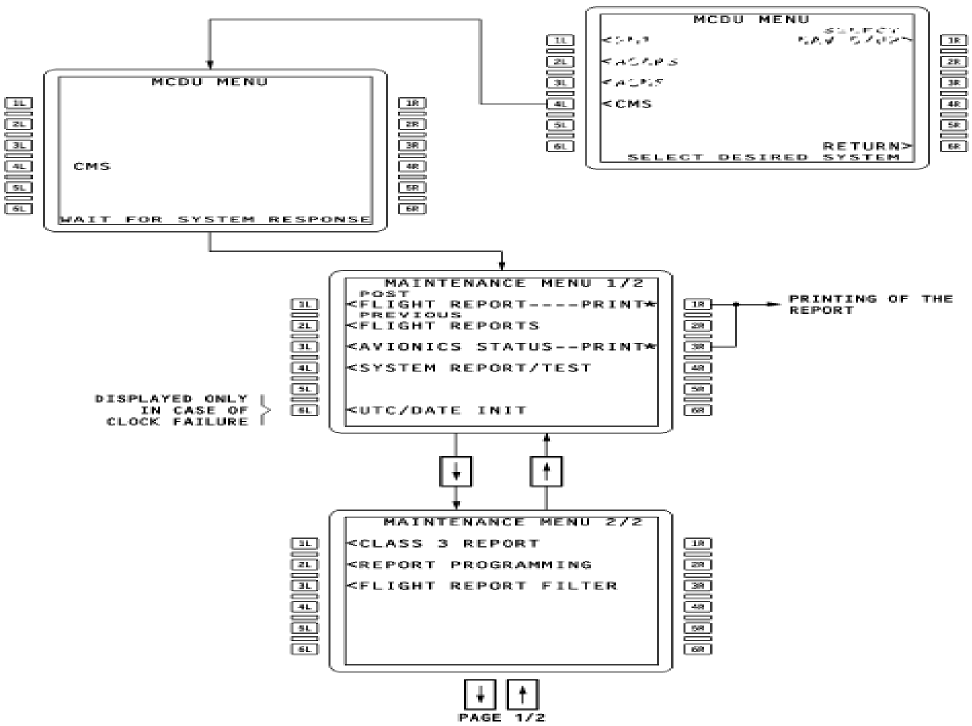

Figure 3-2 On-ground report generation menu (Airbus Industry, 2010c)

3.2.2 Central Maintenance System General Description

Before any mention is made regarding the generation of maintenance reports, it is the author's belief that a concise description of the overall system is necessary. Considering the CMS as an independent system, it could be said to consist primarily of a computer. This computer is called Central Maintenance Computer (CMS) and it is the heart of the system. However, this computer cannot stand on its own, and in order to be functional and provide the ulterior goal of maintenance enhancement it requires system interfaces (Figure 3.3).

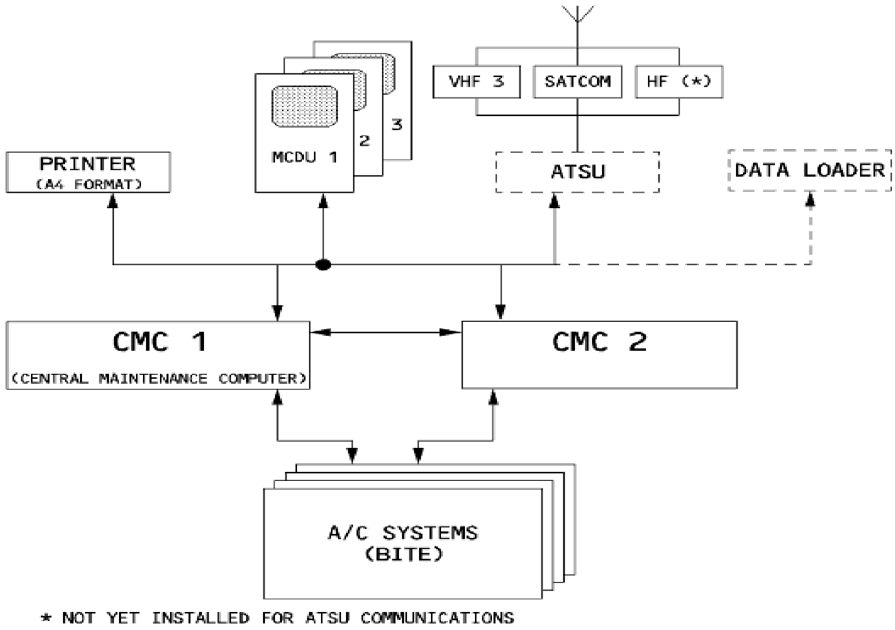

Figure 3-3: Central Maintenance System general arrangement/interfaces (Airbus Industry, 2010c)

It would make sense to divide these interfaces into two; the system interfaces and the human interfaces, with the CMS as the middle connection point. The system interfaces consist of every BITE that is connected to the CMS, as has previously been described. On the other hand, the human interfaces consist of everything that displays the maintenance data and is visible to a human, or that enables a human to control and operate/interrogate the system. In this category are the Multipurpose Control and Display Units (MCDUs), the Multipurpose Disk Drive Unit (MDDU) and the printer. Also in this category are the Air Traffic Service Unit (ATSU) and the Aircraft Communication Addressing and Reporting System (ACARS), if installed (Airbus Industry, 2010e).

3.2.3 Report Categories

The information necessary for maintenance is included in the reports generated from the CMS. It is clear that without them there would be no difference in the aircraft maintenance process. Whatever the kind of the report, it reaches the human operator as a piece of paper or as a message on a screen. Essentially the content of these messages consists of the fault and warning messages that occurred during the flight. By knowing what has been damaged, and where it is located, the maintenance engineers are able to reduce the maintenance time to the minimum, and also to be as accurate as possible in carrying out the maintenance. An example of the content is the ATA reference number, the faulty LRUs part numbers, the time that the fault was recorded, the systems that were affected, and further guidance about the maintenance manual documentation regarding the specific faults that occur each time.

Regardless the content, there are different kinds or report, the main distinction between them related to the means by which they are interrogated by the operator in order for the reports to be generated. Hence we find the reports specifically connected to the CMC and those specifically connected to every BITE for more detailed interrogation. After that, the primary factor that distinguishes them concerns the condition of the aircraft, i.e. whether it is on the ground or in flight. This in turn determines if and where the report will be printed or transmitted.

The category of reports that are dedicated to the CMC begins with the Post Flight Report, which refers to the last flight that the aircraft completed. This report contains all the warnings and fault messages that were generated for the last flight, regardless of whether or not they appeared on a screen. This kind of report can only be transmitted to the ground and printed after the last engine has been shut down. The same applies for the Previous Flight Report, the difference being that the warning and fault messages for the second type could be from the previous 64 flights, and it is limited to the total of 252 messages, unlike the Post Flight Report, which is unlimited but refers only to the last flight.

The Current Flight Report is virtually the same as the Post Flight Report, the only difference being that the former can be transmitted while the aircraft is still in the air, and includes all the warnings and faults that occurred up until the time it was sent. Only two other reports can be transmitted while the aircraft is still in flight; the Real Time Failure and the Real Time Warning, both at the discretion of the operator. The value of these reports could be of great significance on many occasions. It is not a rare event for a flight to be delayed because of aircraft problems. By transmitting important issues to the ground base the maintenance crew has the chance to prepare itself and to assemble the necessary tooling in order to respond more effectively to a difficult upcoming failure.

The Avionics Status Report shows whether and how the systems of the aircraft have been affected by Class 1 and 2 faults. These types of warnings and faults are explained later in this chapter. It cannot be transmitted, but it can be printed both in flight and on the ground. For Class 3 faults there is a dedicated report which amalgamates all the Class 3 faults which occurred in all the systems. It can be printed in flight or on the ground, but there is no option for real time transmission. This report exists in both CMC dedicated reports and individual BITE reports.

Last but not least is the SYSTEM REPORT/TEST. This belongs to the CMC reports, but also constitutes a separate category on its own, and is specific to every BITE. The purpose of this report, which is part of the main maintenance menu, is to ensure a dialog on the ground between the MCDUs and each one of the systems (Type 1, 2, 3) connected to the CMCs.
These reports provide a more detailed interaction between the maintenance crew and each of the systems, and depending on the kind of the system (Type 1, 2, 3) to provide more specific details.

A greater number of reports relate to the Type 1 systems, and their significance is highly critical. Such a system is the Full Authorized Digital Engine Control (FADEC) and the available extracted reports are listed below. All of them constitute the second category of individual BITE dependant reports. For the Type 2 and 3 systems the available extracted reports are reduced successively.

For Type 1 systems, the following reports can be extracted:

- LAST LEG REPORT
- PREVIOUS LEG REPORTS
- GROUND REPORT
- TEST
- LRU IDENT
- TROUBLE SHOOTING DATA
- CLASS 3 FAULTS
- GROUND SCANNING (optional)
- SPECIFIC DATA (optional)

The content of the reports described above is the same as that of the CMC reports, but the information is presented at a more analytical level. The Last Leg Report contains the internal and external faults of the system, and identifies the number of the faulty component, the day and time, as well as the ATA reference. The part number is also available from the LRU IDENT report. The same is true for the Previous Leg Report, but for more than one flight leg. Only Class 1 and 2 faults are shown; for Class 3 faults, there is a separate Class 3 Report.

However the report that maintenance engineers most usually ask for is the Ground Report, which is exactly the same as the Last Leg Report and the Ground Scanning Report, with the option that they can see the computer as it appeared during the flight, and take down the fault messages. The number of the faulty LRU is defined, and the operator can be transferred to the TROUBLE SHOOTING DATA for any supplementary information.

The option of filtering is available in almost all of the reports. This means that the operator can program the filtered parameters by inputting the specific warning and fault messages into the system in order to extract them from the flight. It is also possible to program the time at which the report will be printed or transmitted. The filtering criteria can be printed in a separate report called Flight Report Filter, which also belongs to the CMS dedicated reports.

Except for the reports that the system generates in order to assist the maintenance process, there are those whose content refers to internal CMS functions and affect its fidelity. One such report is the UTC/DATE INIT report, which provides data of time, date etc., in order to reset the system after a power loss. This kind can be generated in flight or on the ground (Airbus Industry, 2010e).

3.2.4 Report Acquisition and Transmission

As has been discussed above, man-machine interfaces are responsible for generating and distributing the reports, according to the kind of the report. The most important of these modular interfaces is the MCDU (Figure 3.4). It is clear that there are two parts to this device; the control and the display. As a display device it behaves in a non-interactive way, in that the operator can see the reports without having to make a hard copy. The option to show the report to someone else via the MCDU is available both in flight and on the ground.

However, the control part of the device is the most useful, enabling a dialog between the operator and the desired interface at each time. Each dialog has a different result in the report. Dialogs between the operator and the CMC provides reports specific to the CMC, but those between the operator and each of the systems gives reports specific to every system's BITE. The generation or acquisition of each report can be either manual or automatic, and both are programmable from the MCDU. In both manual and automatic generation, the printer (Figure 3.5) can be used to make the reports in hard copy.

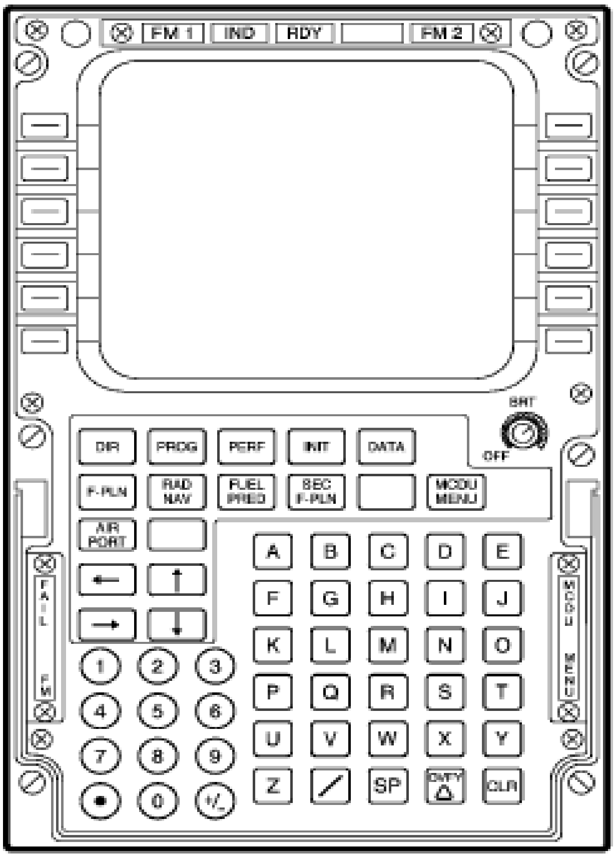

Figure 3-5: MCDU (Airbus Industry, 2008)

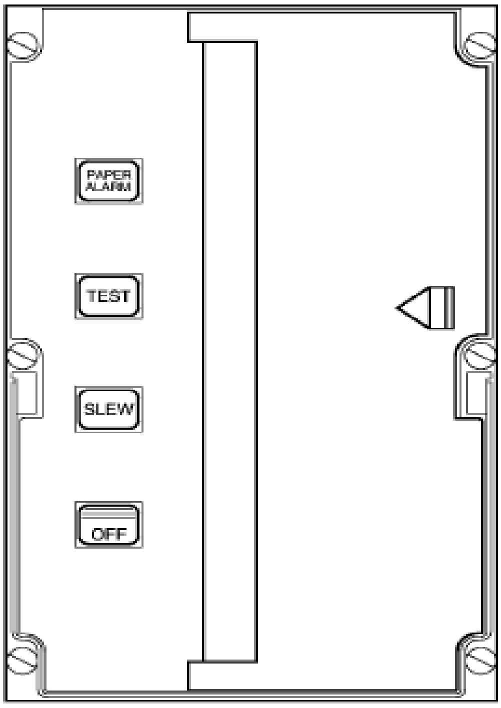

Figure 3-4: Report Printer (Airbus Industry, 2008)

Of particular interest is the transmission of these reports via ACARS, the timing of which is programmed from the MCDU. Transmission is possible both during the flight and on the ground, depending on the type of report, as explained earlier. As has been noted, not all of them can be transmitted from the air. In manual mode the report is sent by the flight or the ground crew to ground stations, while in automatic transmission, internal logics define when a report will be sent. The logics are programmed from the operator through the MCDU. The outcome in every case could be either as a screen display or as a hard copy from printers on the ground. To summarize, the transmissible reports are:

- CURRENT FLIGHT REPORT (in flight)
- REAL TIME FAILURE (in flight)
- REAL TIME WARNING (in flight)
- POST FLIGHT REPORT (on the ground)
- PREVIOUS FLIGHT REPORT (on the round)
- CLASS 3 REPORT (on the ground)
- BITE report in SYSTEM REPORT/TEST mode (on the ground)

Taking as an example a Previous Flight Report, Figure 3.6 shows the print menu, while Figure 3.7 shows the sent menu, both in manual use (Airbus Industry, 2010d).

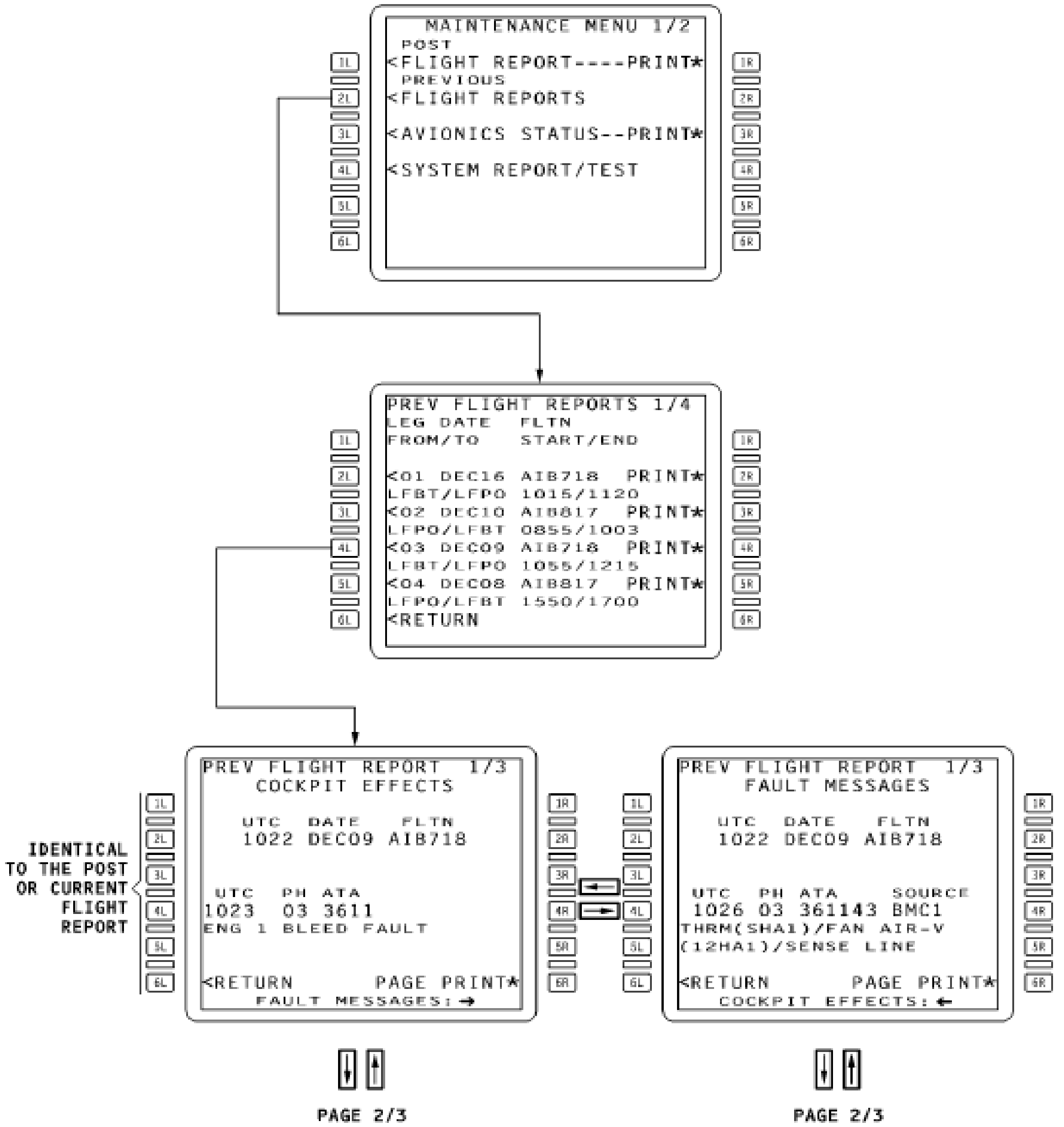

Figure 3-6: Previous Flight Report Print Menu (Airbus Industry, 2010d)

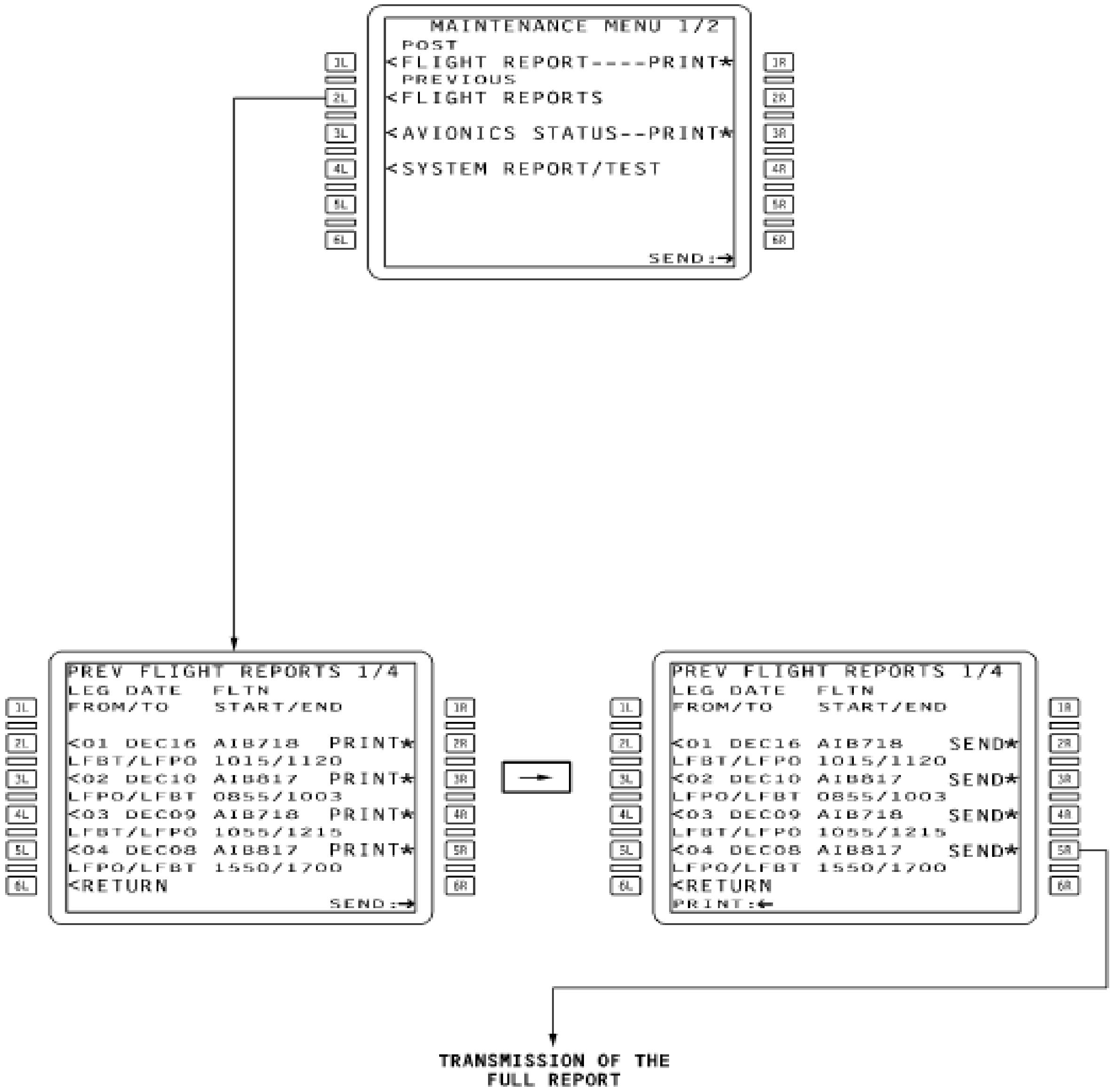

Figure 3-7: Previous Flight Report Sent Menu (Airbus Industry, 2010d)

3.2.5 *Warnings and Faults*

The corrective actions that will be part of the reactive maintenance process of an aircraft begin with the indication for the particular system that there is an error. After receiving the error message, the maintenance engineers take over to restore the system to its previous condition. Not all the recorded errors have the same level of importance, and as such, they are not all useful in terms of maintenance, although that does not imply that they are useless for the flight crew. Regardless of whether they will be used for maintenance purposes or for flight operations, the recorded errors are divided into warnings and faults. Clearly, there are differences between warnings and faults, both in terms of the level of importance, but also the source from which they are generated.

In terms of maintenance, the warnings are less important. The Flight Warning Computer is mostly responsible for the transmission of warnings but it up to the operator which of them need to be recorded to the CMC by using the option of filtering. This is supported by the fact that there is a distinction even within the warnings, between more and less important warnings. The CMC accepts only the primary warnings and the maintenance status type warnings. Furthermore, only the first of the same set of messages is reported, and the less important warnings are transmit to the ECAM.

Conversely, faults are of greater importance and are reported from every system's BITE under the DC2 and DC1 maintenance phases (Duty Cycles). Failures are categorized into three classes. Class 1 failures are the most serious and their effects will have a consequence in the current flight. The numbering from one to three shows the level of seriousness of the failure. Class 2 failures are those of medium importance. They may not affect the current or subsequent flight as long as there is only one failure in the system. Finally, Class 3 failures are the least important, and the aircraft's safety does not depend on them. All three are summarized in the table below.

Table 3-1: Summary of Failure Classes (Source: Author)

FAILURE CLASSES	1	2	3
OPERATIONAL CONSEQUENCES ON THE CURRENT FLIGHT	YES	NO	NO
DISPATCH CONSEQUENCES	REFER TO MMEL MAY BE "GO", "GO IF", "NO GO"	MMEL NOT APPLICABLE REFER TO TSM INTRODUCTION "GO" WITHOUT CONDITIONS CAN BE DEFFERED FOR 800 FLIGHT HOURS	MMEL NOT APPLICABLE NO FIXED TIME QUOTED FOR CORRECTION
INDICATION TO THE FLIGHT CREW	YES AUTOMATICALLY DISPAYED IN REAL TIME WARNING OR CAUTION MESSAGES ON ENGINE/WARNING DISPLAY FLAGS ON PRIMARY DISPLAY OR NAVIGATION DISPLAY OR SYSTEM DISPLAY	YES MANUALLY DISPLAYED ON GROUND (ENG NOT RUNNING) FROM STATUS CONTROL MENUES	NO
INDICATION TO THE MAINTENANCE TEAM	YES AUTOMATIC PRINT OUT AT THE END OF EACH FLIGHT: FAILURE MESSAGES ON THE CMC POST FLIGHT REPORT		YES MANUALLY DISPLAYED: FAILURE MESSAGES ON CMC CLASS 3 REPORT

Only in the case of FADEC, some cockpit equipment gives no indications of faults or warnings. These faults belong to an independent category called Class SM (Scheduled Maintenance) and are corrected during every scheduled maintenance activity. It should be noted that faults are categorized as being either external or internal. This refers to failures on systems components that are shared with other systems. The faulty component would be internal for only one system. For example, the data from one sensor are processed by various systems; however this sensor is an internal part of a single system.

Both warnings and faults are filtered with the filtering function taking place before the POST FLIGHT REPORT is printed, and before every real time transmission. The maintenance crew can activate the function and also upload the filtering criteria to the system via the MCDU. There is an option for no filtering, however, when the filtering is activated, and this applies to all the categories of reports. The last action of the CMC in order to ensure the fidelity of the reports is to correlate the faults and the warnings. There are two types of correlation; faults correlation and warning faults correlation, both of which use the same logic. When the same ATA ref occurs more than once, the system correlates them. The difference is that for the first category the CMC triggers a one minute correlation and for the second, a two minute correlation (Airbus Industry, 2010c).

3.2.6 Central Maintenance Computer Description

The OSA-CBM layers described above appear in the CMS. The first and last layers are covered in the CMS by the human interfaces and the system interfaces respectively. However, the core functions of the system are the responsibility of the Central Maintenance Computer (CMC). As the core of the system and the link between the two kinds of interface, the CMC has to perform actions necessary in order for the system to be useful. These actions are not as important as the report generation which was analysed previously, but they are necessary in order to complete the report with the necessary complementary information, and to provide information about the condition of peripherals that affect the system.

To begin with, the CMC is responsible for the constitution of two very important phase categories, namely the maintenance phase and the flight phase. There are ten flight phases, and the identification of each is achieved by means of collaboration with the Flight Warning Computer that sent the phase to the CMC. This is resumed cyclically every second. In the event of wrong values, the system keeps the last valid value. Nevertheless, before the aircraft is in flight it is crucial for the system to know whether the aircraft is in flight or on the ground. Therefore the CMC has to define the condition of the aircraft in collaboration with other systems before it proceeds to further actions. Essentially, the difference between these conditions is the way that the data are processed, transmitted and stored.

Date and time data are also useful, and for this reason the CMC is synchronised with the clock in order to link every BITE with the exact time and date. Every second it performs acquisition of time and date from the clock. Moreover, a series of flight data is assimilated in the CMC. The CITY PAIR is processed every three seconds, while the flight number is processed every five seconds. These data come from the Flight Management Guidance and Envelope Computer. The A/C IDENT is also taken from the Flight Data Interface Unit every four seconds.

Finally, two more processes are undertaken by the CMC. The processing of the aircraft configuration informs the CMS about the optional systems that are already installed on the aircraft, while the CMC-FDIU-DMU warnings that refer to Class 2 failures or warnings are processed every second.

In addition, in order to ensure the reliability of the system the CMC has its own computer management system. The tasks that belong to this function do not affect the peripheral systems, as they are intended to ensure the internal condition of the CMC. Specifically, the computer performs activity management by checking the coherence between the software and the computer. In addition, it determines the current operation condition, namely master or slave. In the event of a power cut, it can reset itself.

A self-test function enables the system to monitor its own correct operation, which classifies it as a BITE system. If any faults are detected during these tests, the system responds in various ways, depending on the type of failure. The most common action is to switch from CMC 1 to CMC 2 in order to prevent itself. Under normal conditions, CMC1 acts as master. It is also the master computer when a dialog between a human and the system is being performed. In the switching option the system decides how the two CMCs should be engaged (Figure 3.8). Finally, the organization and management of the maintenance data are crucial tasks within this category. Specifically, after the start and end of every flight leg, the CMC manages the files that report faults and warnings (Airbus Industry, 2010b).

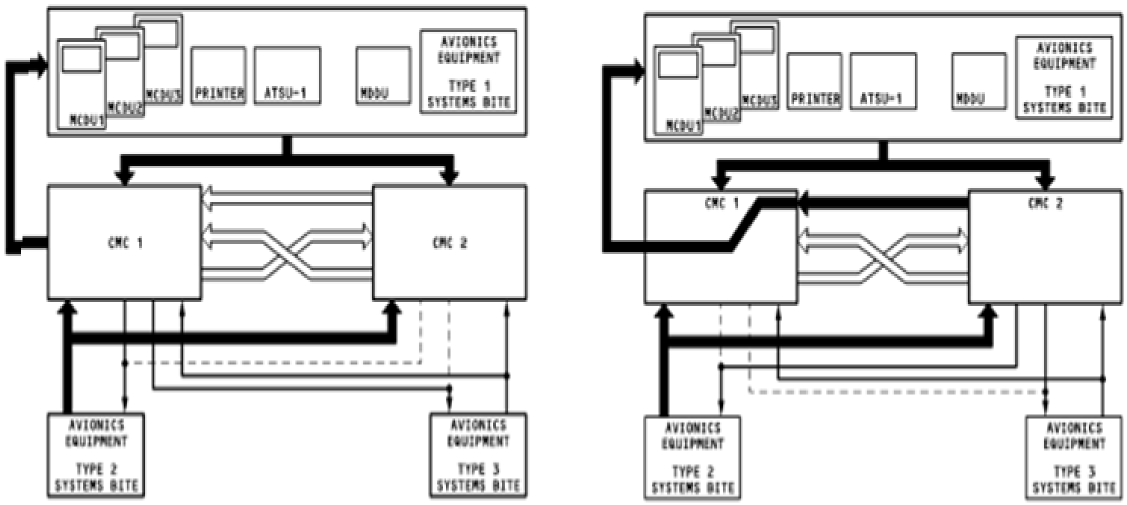

Figure 3-8: CMC Possible Configuration Both Available (left) Switching (right) (Airbus Industry, 2010b)

3.3 Full Authorized Digital Engine Control (FADEC)

The aircraft as a whole consists of many individual systems and within those systems, further components. Even though the general concepts of maintenance apply to all of them, it would be impossible to analyse each of them separate in one study. For this reason the present study focuses solely on the aircraft engine. This choice was not made arbitrarily, but because the engine is considered to be one of the most critical systems on the aircraft, even if it is logical and correct to be consider as in its entirety. However, the main reason was the fact that for aircraft turbine engines, condition monitoring systems that work as maintenance enhancement systems have already been developed and installed.

Therefore, the description of the Full Authorized Digital Engine Control (FADEC) is absolutely necessary as it is the engine's control system, and its functions are aligned with the condition monitoring of the engine, and hence with the Aircraft Condition Monitoring System which follows. To begin with, FADEC is responsible for providing the full range of engine control for all of the phases of a flight. Specifically, it provides gas generator controls, flight deck indication data, engine limit protection, power management, thrust reverse control, feedback, automatic engine starting and fuel return control for IDG cooling (Lufthansa Technical Training, 1995).

In order to operate, FADEC is equipped with an Engine Computer Unit (ECU), which is the core of the system. It consists of a computer which is divided into three parts; a microprocessor for basic and main control functions, and two microcontrollers, one for the communication interface and the other for the pressure transducer interface. Each of these parts uses one of the two existing channels, while the other is for back-up, although the two channels communicate with each other via a crosstalk.

The signals that the ECU sends and receives are also divided into three categories. The first of these signals are the analogs between the ECU and the Throttle Control Unit, which are transmitted to define the Throttle Resolver Angle. Next come the discrete signals, which are transmitted to the ECU from the MASTER LEVER and the A/THRUST LOGIC. The engine ID parameters are also sent with the discrete signals. The discrete outputs refer mostly to Ram Air Turbine extension.

The third category is the digital signals. Several computers that cooperate with the ECU send and receive signals for different purposes. The main inputs come from the Air Data/Inertial Reference System and the Engine Interface Unit, while the outputs go to the Engine Interface Unit, as well as the Flight Warning System, Display Management Computers and the Flight Management and Guidance Computer. Constant interaction is necessary between the Flight Management and Guidance Computer, Engine Interface Unit and ECU. All of the digital signals are transmitted via ARINC 429 data busses.

However, the most important data, the acquisition of which is necessary for any kind of control and also for any attempt at condition/trend monitoring, are the digital outputs from the ECU, collected from the engine itself. These signals are the initial point of the condition monitoring process, and sensors for their acquisition are mounted in various positions in the engine (Figure 3.9). As inputs to the ECU they could be discrete as well as digital, but as outputs they have to be digital. The former verifies the OSA-CBM architecture. They provide constant measurements of temperature, pressure and speed, and the data, which are processed in the ECU, provides indications, alerts and warnings for the engine modules, engine condition monitoring data and FADEC system maintenance data utilizing the BITE. In some cases, the computer also provides suggestions for optimum handling, and can perform such optimum handling on its own, without any human intervention.

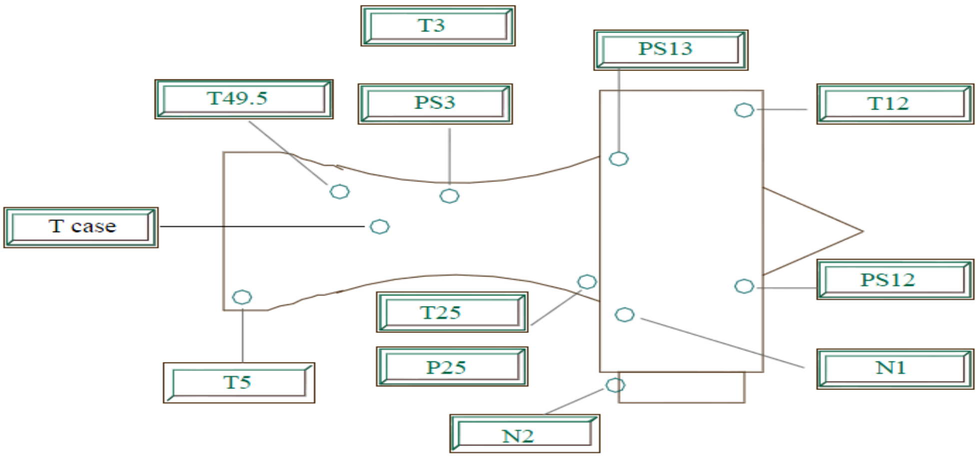

Figure 3-9: Engine Monitoring Sensors Mounted Positions (Lufthansa, 1995)

Once these data have been obtained, the cycle of the actual interfaces is complete (Figure 3.10). Different systems might use these signals for different purposes, but there are only two possible destinations for them. The first is the ECU where they are processed. In terms of maintenance, the ECU meets as a BITE system by performing the functions of monitoring, self-testing and faults detection, isolation and memory.

The second receiver of these signals is the Engine Interface Unit, also called Engine Interface and Vibration Monitoring Unit (EIVMU) by many manufacturers. Initially the EIVMU is the concentrator between the aircraft and the FADEC of every engine. It is necessary for every engine to have its own EIVMU in order to achieve engine-to-engine segregation. The signal transmission from the engine is received by this unit, where it ends. It is EIVMU's responsibility from now on to decide how to manage these signals. As the agent between engine and the aircraft, the EIVMU has to 'feed' the engine with all the necessary data in order to operate. Furthermore, the various aircraft systems take from the EIVMU information about the engine's condition. Additionally, it provides the ECU with a power supply. Just like the ECU it has two independent channels and can transmit and receive analogue, discrete and digital signals. Figures 3.11 and 3.12 show the inputs and outputs respectively (Lufthansa Technical Training, 2007; EASA, 2005).

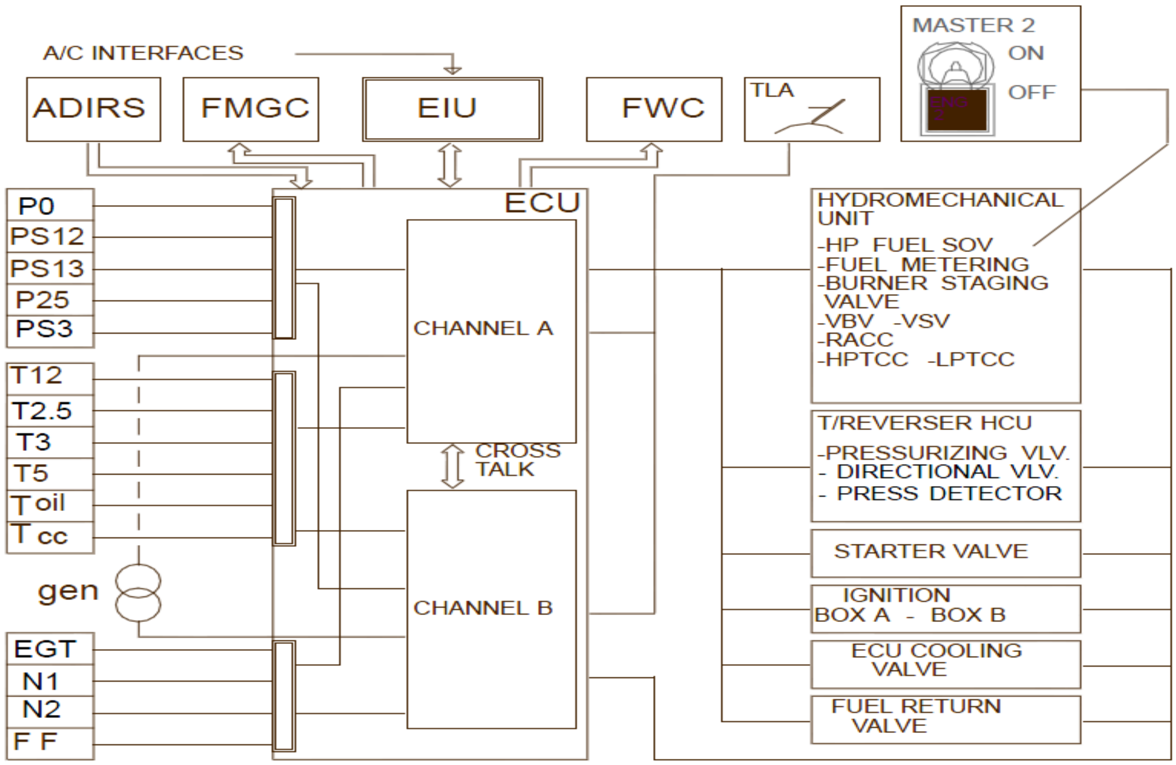

Figure 3-10: EIVMU Interfaces (Lufthansa, 1995)

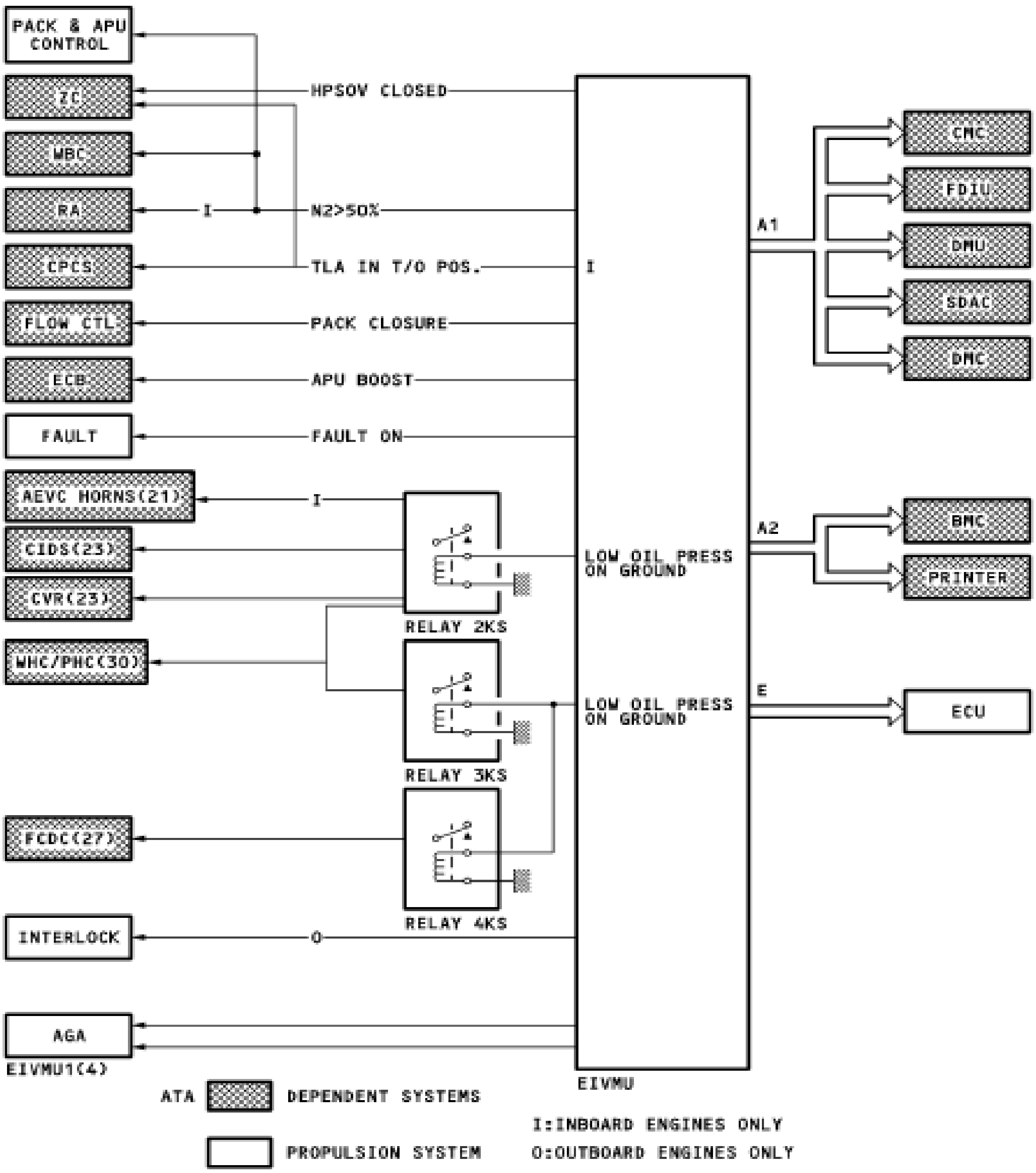

Figure 3-11: FADEC Outputs (Airbus Industry, 2010a)

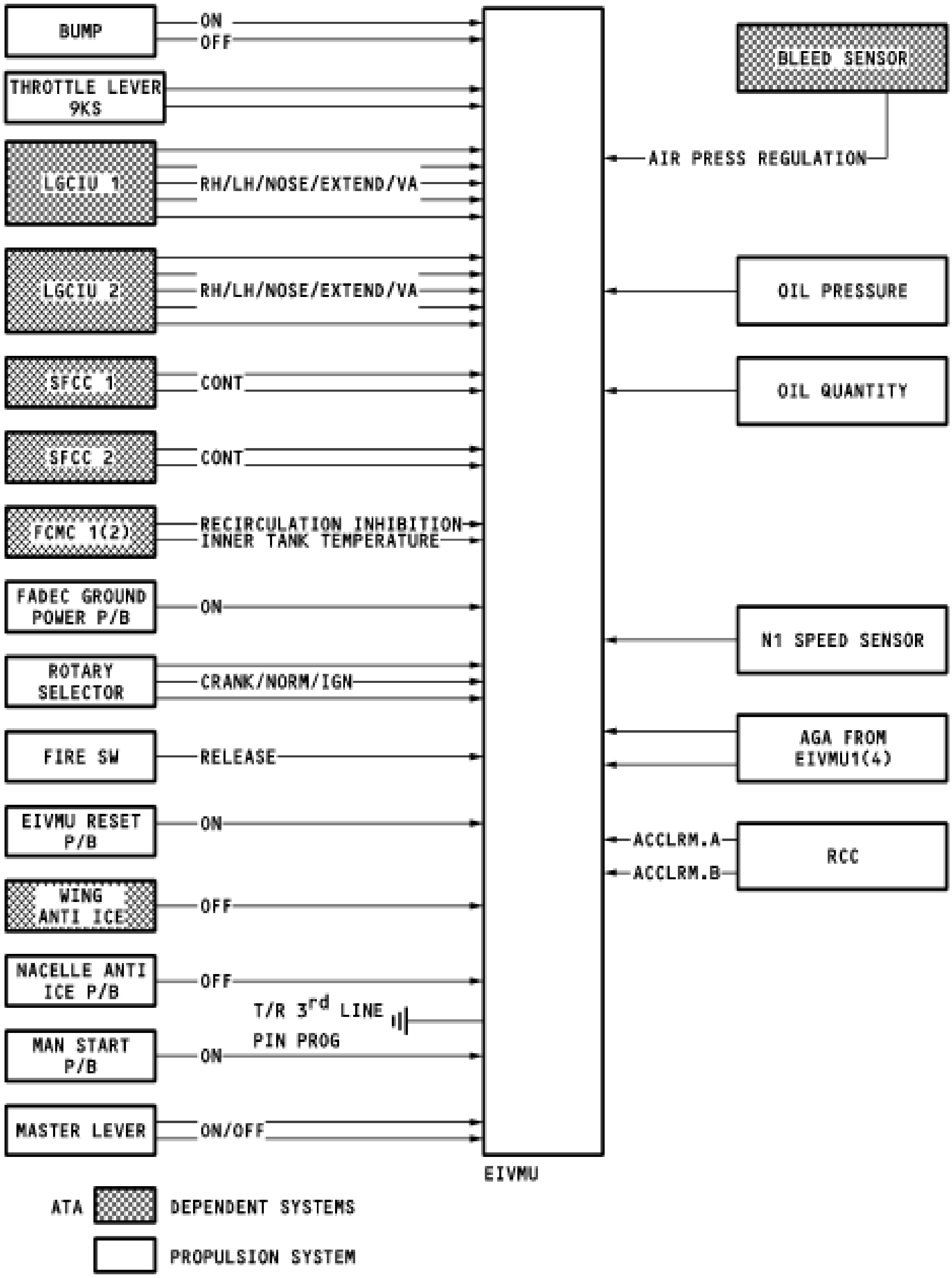

Figure 3-12: FADEC Inputs (Airbus Industry, 2010a)

3.4 Aircraft Condition Monitoring System

3.4.1 Purpose and Interfaces

In comparison with the reactive nature of the CMS, preventative/predictive maintenance is implemented in modern aircraft by means of the Aircraft Condition Monitoring System (ACMS). Although its name suggests that it includes the entire aircraft, its functions mostly concern the condition monitoring of the engine. After the CMS, this is the second system for maintenance enhancement on the aircraft, and its general functions are similar to those of the former. It is used not at line maintenance level, but for more long-term aircraft monitoring. The purpose of the monitoring has to do with the maintenance activities that are going to be performed on the aircraft. The main point of the system is to turn the traditional fixed interval maintenance to on-demand maintenance, and hence enable more accurate replacement by means of time and better logistic organization by exploiting the prognostic capabilities. Moreover for the monitored systems the benefits can take the shape of avoidance of expensive unscheduled and unanticipated maintenance actions outside the main base of the aircraft operator.

In order to function as a system, the ACMS needs two things. The first is the diagnostics/prognostics method that best applies to the monitored component and the second is the system interfaces. The diagnostics/prognostics methods of the second chapter could be profitably implemented on such a system concerning an aircraft engine. The data that are necessary are those that are manipulated by the FADEC, described previously. On the other hand, the interfaces that enable the crew to operate the system are the same as those of the CMS; specifically, MCDU, the ACARS, the printer, and the MDDU. The only exception is that for this system the operator has the additional option of taking the generated report or the monitoring data on a floppy disk from the MDDU.

In addition to the human interfaces, the system is equipped with a series of components that enable it to perform its tasks. Starting with the less important ones, the system has an optional Digital ACMS Recorder (DAR). The function of this, clearly, is to record data in order for use in ground activities. The recording is done onto a rewritable magnetic tape with a storage capacity of 128 Mbytes per disk. The data are transmitted from the DMU in HARVARD BIPHASE, normally at a rate of 128 words per second. The process of DAR engagement could be either manually from the MCDU, or automatic.

An alternative option for data storage is the Smart ACMS Recorder, which is integrated into the Data Management Unit (DMU). The data are stored in a non-volatile Solid State Mass Memory, with limited capacity. The use of very sophisticated algorithms guarantees the most efficient use of the memory module. The operator retrieves flight data on a floppy disk from the MDDU, while the Smart ACMS Recorder data are stored internally in the Solid State Mass Memory and extracted through the MDDU. The SAR creates files for the data and monitors and stores them in a specific file. Normally for condition monitoring it creates one file for every flight leg. Additionally it has 8 independent recording channels with a capacity of 127 parameters per channel. It is up to the operator to decide which of these recording systems to use, and he can program the recording triggering for both of them.

The most important part of the system is the DMU. This is the core of the system, and the component which coordinates the internal interfaces of the system (but not the human interfaces) in order to retrieve data and process them. However, communication for the human dialog interfaces is provided by the DMU, which is a computer able to perform real time processing of the data that are sent to it from several modules. All data can be stored in the recorders mentioned above and distributed to the different systems as required. The operator can control a number of DMU functions via the MCDU, namely the parameter label call up menu, parameter alpha call up menu, special functions/reprogramming submenus, list of previous reports, list of stored reports, manual request of reports, list of stored SAR data, manual request of SAR recording, and manual DAR recording control.

The data for processing comes to the DMU from the various systems that are connected to it. All these systems could be considered to be the intake interfaces, or the parameter sources of the DMU because they provide but do not take data but from the DMU. One of the systems that provide information is the EIVMU, which is the primary focus of this research, because it refers specifically to the engine (Airbus Industry, 2010a). The modules discussed above complete the description of the interfaces of the DMU (Figure 3.13).

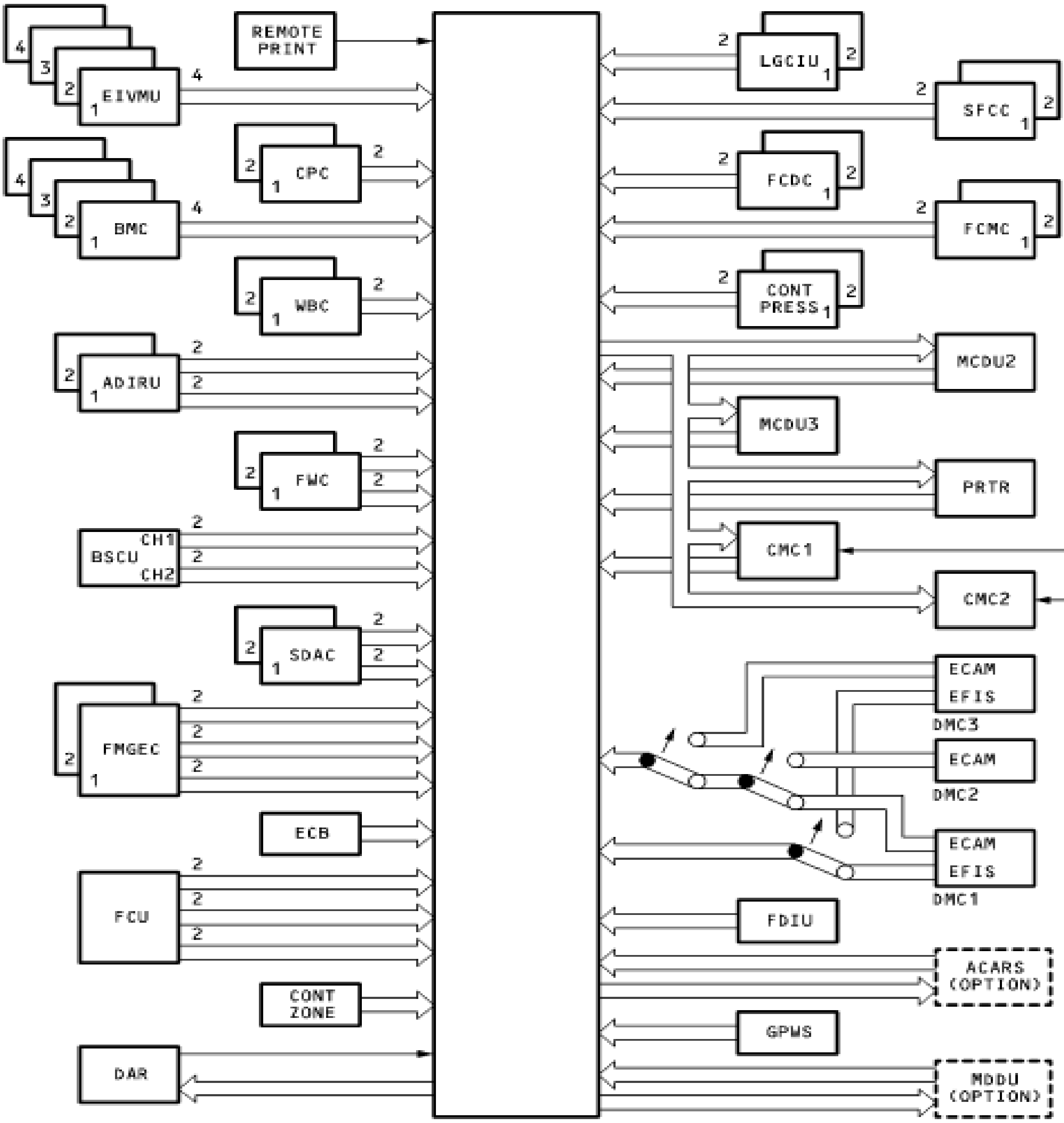

Figure 3-13: DMU Interfaces (Airbus Industry, 2010a)

The DMU performs the following tasks, in the following order:

- External parameter data sampling and validity checking
- Alternate parameter selection
- BITE Centralized Data conversion
- Parameter filtering
- Evaluation of logic expression required by GCVs.
- Global computed variables (GCV) generation
- Filter parameter data conversion
- Evaluation of logic expression not required by GCVs.
- ACMS report trigger evaluations
- SAR/DAR recording trigger evaluations
- Packed discrete word creation
- History buffer generation
- ACMS report data collection
- DAR data collection and recording

3.4.2 ACMS Reports

The major activity of the ACMS is to create the reports which are the result of the data collection and analysis that the system performs. Like the CMS, this is the most interesting part for the operator because through these, he is able to check the systems that are being monitored. According to the monitored data and the diagnostic/prognostic method that is implemented, the operator makes decisions about the optimum maintenance interval. The reports that the system is initially programmed to generate are specific, and are known as Standard ACMS Reports. They are listed below, and a sample is illustrated in Figure 3.14.

.

- <01> Engine Cruise Report
- <02> Cruise Performance Report
- <04> Engine Take-Off Report
- <05> Engine On-Request Report
- <06> Engine Gas Path Advisory Report
- <07> Engine Mechanical Advisory Report
- <08> Engine Trim Balance Advisory Report
- <09> Engine Divergence Report
- <10> Engine Start Report
- <11> Engine Run-Up Report
- <13> APU MES/IDLE Report
- <14> APU Shutdown Report
- <15> Load Report
- <16,17,18> Programmable Reports (DMU)
- <19> ECS Report
- <20> Ram Air Turbine Report
- <41, 42, 43, 44, 45> Programmable Reports (FDIMU)
- <50> Cruise Performance Stability Statistic Report

Because these are the standard reports generated by the system, all of the conditions that affect them are predetermined; in other words, the data that are collected, the output format and the trigger conditions (the factors that cause a report to be generated) are fixed. Such factors could be fixed time intervals or specific conditions on the system that is being monitored. These could be stability criteria, divergence criteria or general anomalies in the system, such as an excessively high temperature in the engine. The manual generation of a report also constitutes a trigger condition, independent of any other logic circumstance. The manual trigger logic is available via the MCDU, and could be output as hard copy or ACARS downloading. The remote print button can also be used to generate hard copy output.

The operator can use the programmable report trigger to program the parameters to generate a report. The system is able to process more than 400 programmable logics and can store approximately 65 printable reports. In the event of it being overloaded, the oldest report is erased, after being printed or sent to ACARS. The history buffer is 20s for 400 parameters, with an average sampling rate of 4 per second.

Every report contains standard information such as the report number and name. In addition, each report defines the engine hours, engine cycle, engine running time, and flight leg. Engine hours are calculated from the moment the nose gear is decompressed until is compressed again, while engine cycle counts every climb. Data relating to the flight leg is taken from the CMC.

Should the standard report not meet the operator's requirements, it is possible to create a customized report via the Free Programmable Reports menu, which offers the option to create more detailed or sophisticated reports, which may be better suited to the airline's needs. The main characteristics of these customized reports are no different from the classic reports, and the operator can program the report declaration, which specifies criteria such as report title/number, enabling of the remote print button, maximum number of reports of this type to be stored in the print buffer per flight leg, maximum numbers of reports of this type to be transmitted through ACARS per flight leg, number of flights after a report is triggered, and generation intervals during flight. Moreover, the trigger conditions, the layout format and the report processing algorithm are also programmable.

The final functions to note are those performed by the Ground Support Equipment (GSE). These are ground-based software applications used for displaying the information that has been collected, and are used for reprogramming certain aspects of the ACMS functions, or for generally downloading and uploading data or configurations. The computer used on the ground is a DMU external IBM-compatible personal computer. Creating and editing of the

ACMS setup database is provided for DMU upload. A history function is provided, as well as a listing of which aircraft has which DMU setup database version. Nevertheless, the fact that the GSE does not share restrictions that apply on board, such as module space, computational and storage requirements etc., are used for further analysis of the data obtained. In other words, the data are stored there, and a further more detailed processing of them is possible and doable (Airbus Industry, 2010a).

```
1234567890123456789012345678901234567890123456789012345678901234

                 <Free programmable 64 characters>
                 <Free programmable 64 characters>

                 A340 ENGINE TAKE OFF REPORT <04>     PAGE 01 OF 01

   ACID     DATE     UTC       FROM TO    FLT         CODE CNT

C1 XXXXXXX 99AAA99 99.99.99 AAAA AAAA XXXXXXXXXX 999X 999 ..

   PRV PH   TIEBCK DMU IDENTIFICATION   MOD AP1 AP2

C2 099 99.9 XXXXXX SXXXXX VXXXXX CXXXXX XXX 999 999 ..

   TAT   ALT   MN    SYS (....... BLEED  STATUS .......) APU

C3 X99.9 X9999 0.999 999 9.99 1111 1111 1 1111 1111 9.99 1 ..
C4 X99.9 X9999 0.999 999

   ESN     EHRS  ERT   ECYC  SOURCE STATUS    SR ECW1  ECW2  NM

C5 999999 99999 99999 99999 XXXXXXXXXXXXXXX XX XXXXX XXXXX X  ..
C6 999999 99999 99999 99999 XXXXXXXXXXXXXXX XX XXXXX XXXXX X  ..
C7 999999 99999 99999 99999 XXXXXXXXXXXXXXX XX XXXXX XXXXX X  ..
C8 999999 99999 99999 99999 XXXXXXXXXXXXXXX XX XXXXX XXXXX X  ..

   T/O DELTA N1 SUMMARY

N1 99 99 99 99 99 99 99 99  ..

   N1      N1C    N1MX  N1A     N2     EGT   FF     P0    T12    P25

S1 999.99 999.9 999.9 999.99 999.9 X999 99999 99.9 X99.9 99.99 ..
S2 999.99 999.9 999.9 999.99 999.9 X999 99999 99.9 X99.9 99.99 ..
S3 999.99 999.9 999.9 999.99 999.9 X999 99999 99.9 X99.9 99.99 ..
S4 999.99 999.9 999.9 999.99 999.9 X999 99999 99.9 X99.9 99.99 ..

   T25   PS3   T3    T5    VSV  VBV  HPT LPT RAC GLE   PD TN

T1 X99.9 999.9 X99.9 X99.9 X9.9 X9.9 999 999 999 999.9 99 X99  ..
T2 X99.9 999.9 X99.9 X99.9 X9.9 X9.9 999 999 999 999.9 99 X99  ..
T3 X99.9 999.9 X99.9 X99.9 X9.9 X9.9 999 999 999 999.9 99 X99  ..
T4 X99.9 999.9 X99.9 X99.9 X9.9 X9.9 999 999 999 999.9 99 X99  ..

   OIP OIT   VF    VC    VH    VL    PHF PHT BBF  BBT  EVM

V1 999 X99.9 99.9 9.99 9.99 99.9 999 999 9.99 9.99 XXXXX  ..
V2 999 X99.9 99.9 9.99 9.99 99.9 999 999 9.99 9.99 XXXXX  ..
V3 999 X99.9 99.9 9.99 9.99 99.9 999 999 9.99 9.99 XXXXX  ..
V4 999 X99.9 99.9 9.99 9.99 99.9 999 999 9.99 9.99 XXXXX  ..
                                                                   Form
                                                                   Feed
```

Figure 3-14: Sample of ACMS Report (Airbus Industry, 2010a)

Chapter 4: Case Study

4.1 Introduction

It should be noticed by now that the evolution, in performing maintenance in the aviation sector, is great. By utilizing the methods and the systems described in the second and third chapter respectively, the outcome could only be beneficial in terms of costs savings, remaining useful life of the material and facilities, personnel and logistics organization. However, improvement could arrive if the condition monitoring capabilities utilized in order to device a maintenance plan before the engine is in service with the only difference that the data needed would be simulated and not real.

As a proof of concept that condition monitoring works beneficial for the maintenance process, a case study is following. The method adopted indicates the Engine Usage Diagnostics as they have the ability to be performed for design stage assessments regarding the maintenance planning but also for post service with more accurate data. The simulated scenarios in the particular research will focus on the impact of the take-off segment on the engine blade’s remaining useful life as the ambient condition and the trust setting will be changed each time. Different condition will indicate different fatigue and hence different maintenance planning as the remaining useful life will depend on the particular operating condition. The research will focus on a paramount failure mode, creep, and the assessments will performed by using creep life analysis.

Because the life assessment calculations for a whole engine are potentially endless, this research focuses on a specific component, namely the high pressure turbine rotor blade. This was chosen because the turbine section is subjected to extreme stress and thermal forces, the combination of which results in an accelerated rate of damage. In other words, this means that the components that are located at the turbine section are the most vulnerable to damage. This is clearly shown in figures (Figures 4.1 and 4.2). For both failures and downtimes, the turbine section, along with the combustion chambers, accounts for the highest percentage. Additionally, the maintenance of the blades which is essentially their replacement, as the restore actions on them could be more expensive and complex, are holding down the whole aircraft. Hence, an accurate estimation of their remaining useful life is of paramount importance in order to avoid extended downtimes but also to combine these downtimes with necessary maintenance for the rest of the aircraft.

This component category is designed and manufactured according to the damage tolerance philosophy. Briefly, damage tolerance philosophy promotes the condition monitoring of the components in order to replace individual components, which is opposite of the safe life philosophy, in which components are replaced at predetermined times because of their criticality (Andreadis et al., 2009). Despite the fact that the turbine blades are considered to be highly critical components, especially for single engine aircraft, it is observed that it is more advantageous from a financial perspective to perform constant monitoring of these components, with its associated disadvantages, than to replace the components at predetermined times, because of the high costs involved (Chamis, 1999).

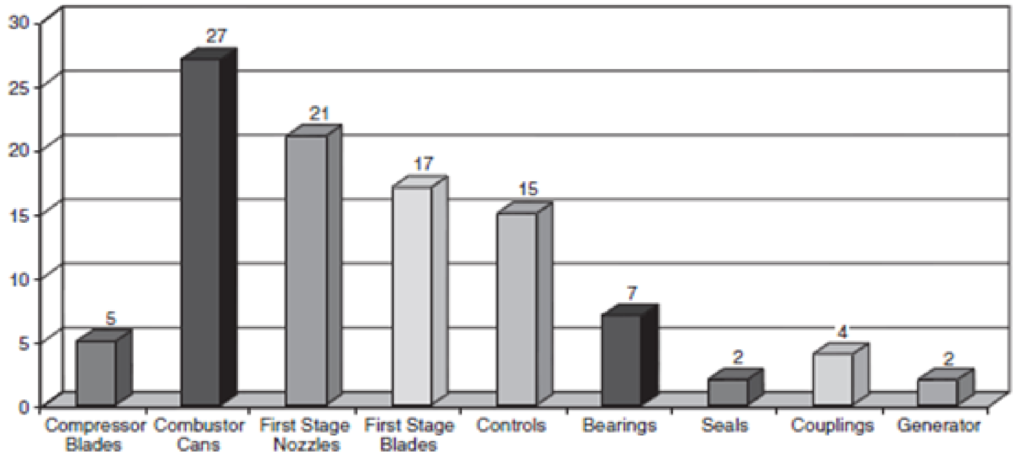

Figure 4-1: Contribution of Various Major Components to Gas Turbine Downtime (Boyce and Referex, 2002)

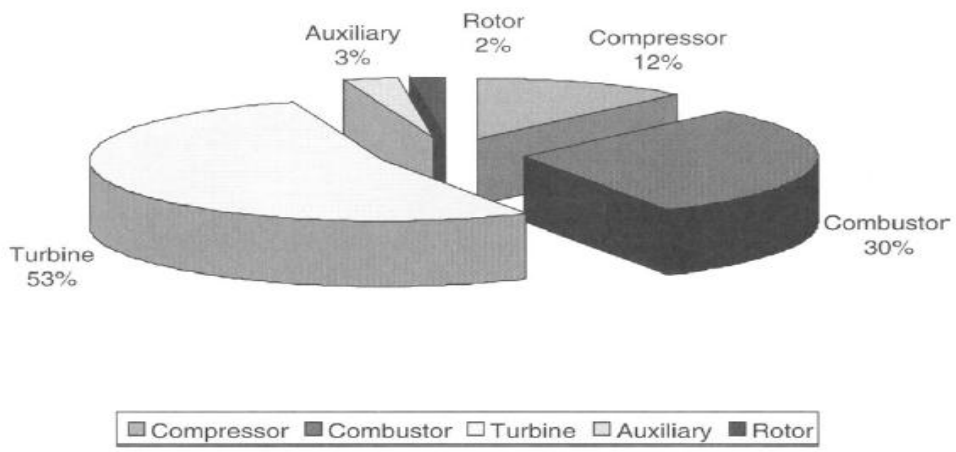

Figure 4-2: Major Failures in Gas Turbines less that 170MW (Boyce and Referex, 2002)

4.2 Engine Description and Modeling

The engine that has been chosen in order to undertake the experiments in this research is a CFM56 5B2 engine (Figure 4.3), which belongs to the CFM56 family and is jointly manufactured by Snecma and General Electric. This particular model is a high-bypass turbofan two-spool engine that produces a total of 31000 lbs (137.9 KN) of thrust at take-off. The engine is of a modular design, thus enabling maintenance to be performed more easily by maintenance workshops having limited repair capability. Modular maintenance is concerned primarily with replacement of modular assemblies and parts. The major modules are the fan and booster, the compressor module, the turbine module and the accessory drive module. This general assembly concept is followed by all the models of the CFM56 family, but each displays different specific performance characteristics depending mostly on the age of the engine and the application to which each engine refers. The specific characteristics of the CFM56 5B2 are shown in Table 4.1 at the end of this section (Lufthansa Technical Training, 1995).

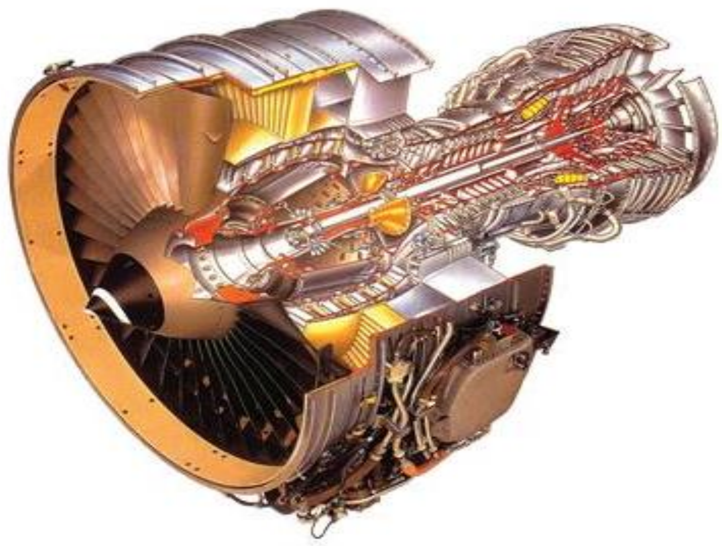

Figure 4-3: CFM56 5B2 Cutaway Drawing (http://www.cfm56.com/products/cfm56-5b/cfm56-5b-technology)

This type of engine is designed specifically for Airbus aircraft, and is installed on the Airbus A-321. This is a medium-range narrow-body jet airliner, which was first introduced in 1993, making it a relatively modern aircraft. This particular type of aircraft is used by hundreds of operators due to its ability to perform domestic, regional, and also transatlantic flights.

Hence, it constitutes a representative contemporary example, which is why this particular engine was chosen for examination (Aircraft Commerce Journal, 2006).

The modelling of the engine was undertaken by means of a performance simulation code, Turbomatch, which was developed at Cranfield University. It is important to note that in the present study, the fidelity of the Turbomatch tool was not taken into consideration, and so the results obtained are considered to be the most accurate and the closest possible to real conditions. As a modelling and computational tool, Turbomatch has the ability to provide results concerning the performance parameters of the engine depending on the operating conditions. These performance parameters concern the sum of the performance data, such as thrust, exhaust gas temperature and efficiencies, but also others such as rotational speed, pressure and temperature. The data provided by the simulation code are identical to those that a true application uses for health management processing, such as the ACMS described in Chapter 3.

An encoded model of the engine is essential for the Turbomatch performance simulation program. This initial model will respond to a specific performance operating point of the engine, with particular given atmospheric conditions. In order for the coded model to be accurate and valid, the simulated data must match the real data for the same specific condition that was used for modelling. In other words, the thrust, turbine entry temperature (TET) and so on, given from the simulation program, must match the real values. This will provide the design point of the simulation code that will later act as a reference point for the other operational conditions. For the current study, the design point of the engine was chosen to be at maximum climb thrust, because all the real data available, such as TET, pressure ratios and efficiencies, were for that operational segment, and so it was possible to validate the results from the simulation code in conjunction with the real one. A number of runs were undertaken until the desired matching was achieved, at which point the simulation was ready.

Once the design point was ready, it was then possible to investigate the behaviour of the engine for different performance and atmospheric conditions by changing them according to the desired scenario each time. Those conditions, different from the initial ones, will be formed by interchanging certain parameters from the design point. This change will eventually change the overall performance model of the engine, which will be then running for off-design points. The present study is concerned with the ISA deviations and the de-rated thrust settings, so these are the parameters that were changed from the design points. A change in one of them will bring about a change to the whole performance model of the engine. For example, the design point of the specific engine being studied was carried out at maximum climb thrust of 6.420 lbs (28.6 KN), and gave specific values for pressures, temperatures, rotational speed etc. If the thrust were to change, then the other parameters would also change. The same would also happen if the change were to occur on an atmospheric parameter such as ambient temperature or altitude, or to anything else that is pre-established in the code (Pachidis, 2005; Cranfield University, 2000).

However, for this version of Turbomatch the two necessary initial inputs, even for off-design runs, are the thrust and the TET, and the absence of either will compromise the results. This means that there is a specific matching of thrust and TET for a given atmospheric condition which has to be known beforehand and be entered as initial input, otherwise the tool is unable to provide results. For example, when the engine is operating at 30000 lbs of thrust, the corresponding TET is 1500k; for 28000 lbs of thrust the TET is 1450k etc. Because the matching of thrust and TET was made piecemeal, there is a kind of deviation between the desired thrust value and that which was finally matched with the TET. For example, the TET at cruise thrust is considered to be 1279.39 K. However that TET was matched for thrust of 25.94 KN with the nominal cruise thrust being at 25.98 KN. Such small deviations exist in all of the code runs, but are considered negligible. Nevertheless, they mentioned in order to be completely accurate with our assumptions. The input code file can be found in Appendix A along with a small explanation of it.

Table 4-1: CFM56 5B2 Technical Characteristics (Lufthansa Technical Training, 1995)

CFM56 5B2 Technical Characteristics	
Max take-off thrust (lbs)	**31000**
Max climb thrust (lbs)	**6430**
Max cruise thrust (lbs)	**5840**
Flat Rated Temperature (^{o}C /^{o}F)	**30/86**
By Pass Ratio	**5,5/1**
Mass Flow (lbs/sec)	**956**
Overall Pressure Ratio	**35.5**
EGT	**950**
N1 (RPM)	**5200**
N2 (RPM)	**15183**
Length (inch)	**102,4**
Fan Diameter	**68,3**
Basic Dry Weight (lbs)	**5250**
Fan Stage Numbers	**1**
Low Pressure Compressor Numbers	**4**
High Pressure Compressor Number	**9**
High Pressure Turbine Stage Numbers	**1**
Low Pressure Turbine Stage Numbers	**4**
Applications	**Airbus A321**

4.3 Creep

Along with the engine characteristics and modelling, it is essential to have a good understanding of the failure mechanism under investigation. Creep, along with fatigue and corrosion, is considered to be the primary cause of failure of turbine blades. This phenomenon could be explained as the time-dependant and progressive plastic deformation of a material/component under the impact of temperature and stress forces. For a turbine blade that deformation is expressed by lengthening of the blade across the direction of the stresses.

The basic principle behind creep is that the grain boundaries are weakened by the impact of loads and stresses in combination with high temperatures, and with the passage of time. Hence, the basic consequence of creep is the degradation of the strength of the material due to the higher mobility of the atoms at greater temperatures. This phenomenon concerns the majority of metallic materials, and the internal problems that might occur are deformations at grain boundaries, such as crystallization and grain growth, and possible destructive oxidation with possible intergranular penetration of oxide, and hence acceleration of the erosion and corrosion failure modes. Figure 4.4 shows the material before and after the appearance of creep.

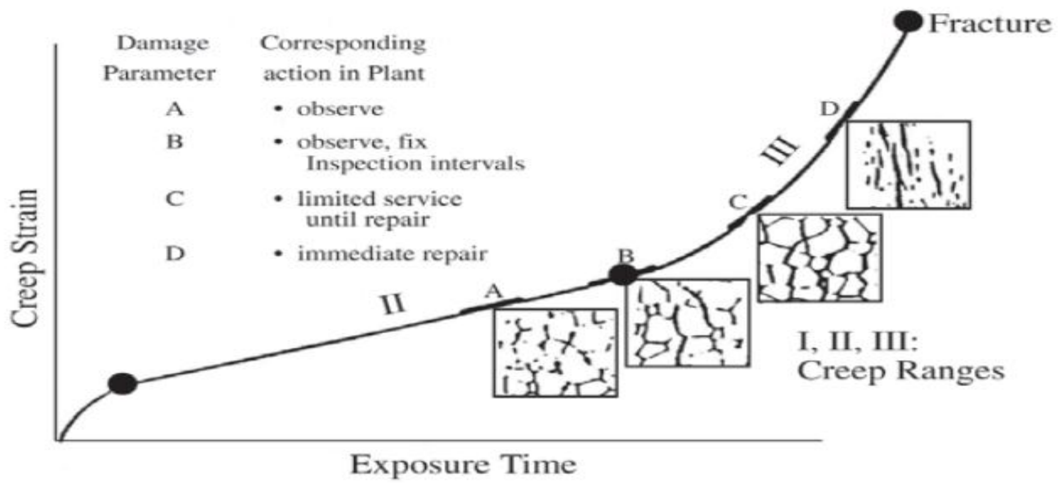

Figure 4-4: Neubauer's classification of creep damage from observation of replicas and consequent action to be taken (Furtado and May, 2004)

The effects of creep are obvious and significant when the operating temperatures exceed 50% of the materials' melting point. Materials have resistance to creep loading, which is described as their homologous temperature, or melting point. The homologous temperature of a material is the absolute temperature to absolute melting temperature ratio. Aero-engines alloys will usually creep at 0, 5*Tmelting to 0, 7*Tmelting temperature (Andreadis et al., 2009).

There are two major creep mechanisms, and the difference between them lies in the different shape of the internal structure of the material. Dislocation creep results from the increased number of dislocations in materials at high levels of stress, and their increased mobility at high temperatures, while diffusional creep is the result of the migration of the atoms from grain boundaries under tension to boundaries which are perpendicular to the direction of the maximum stress and the migration of voids in the opposite direction.

From the initial occurrence of the phenomenon until the final stage of fracture, three stages of creep growth can be observed (Figure 4.5). The first stage is called primary creep, in which strain resistance is observed, due to the fact that the microstructure imperfections of the material have been improved. It should be noted that there is a latent and often overlooked stage before the primary creep, known as instantaneous strain or initial load. It is often overlooked because it is considered to be negligible, and also partly recoverable, or not dependent on time. The second stage is secondary creep, which accounts for the majority of the creep lives of the materials, and in which the creep rate seems to be constant. Finally, tertiary creep is an unreal situation in which creep is accelerated and the material/component is fractured (Cookson and Haslam, 2009). However, it should be noted that after the creep occurrence, the blades need to be replaced, and so the engine will be subjected to maintenance.

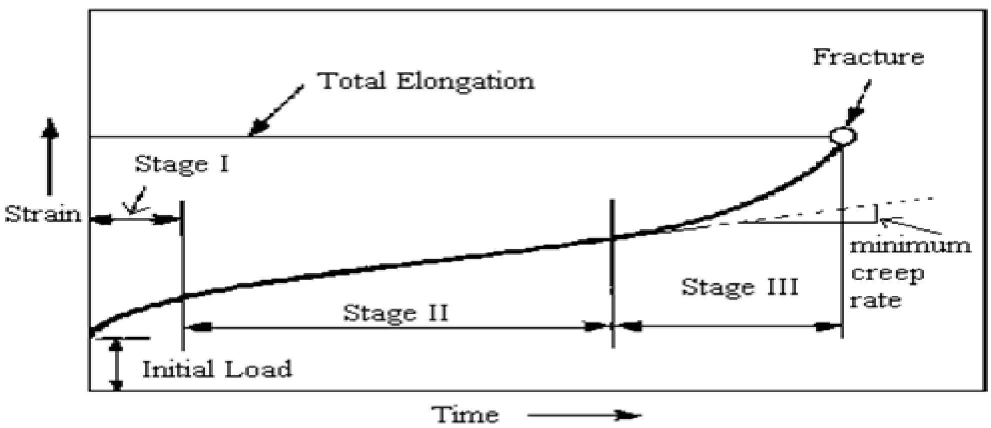

Figure 4-5: Creep Stages (Allison et al., 2010)

Due to the fact that all the factors that encourage creep growth are usual and common for the turbine sections of aero-engines, it is critical to be aware of creep in order to avoid its undesired consequences that compromise both the performance and reliability of the engine. Even though the predictions regarding the initiation of the phenomenon will have already been made before any engine is launched onto the market, constant monitoring of it would still be beneficial. This is because the predictions that the manufacturer provides refer to specific performance and ambient conditions; something which is impossible to always comply with in reality.

Of course, by incorporating a very generous safety factor percentage, the manufacturer avoids the greatest danger of failure during service. However, because this research focuses on maintenance aspects, so the study of creep as a diagnostic method for calculating the remaining useful life of the turbine blade, and thus maintenance planning in order to avoid unscheduled and undesired downtimes is clearly relevant.

4.4 Case Description

As has been made clear in this research, the maintenance activities relating to every piece of equipment on an aircraft are predefined at a high level. This is because the manufacturer creates a maintenance planning document for the particular aircraft, and also for the engines with which it is fitted. However, the main disadvantage of this maintenance plan is that it can only be efficiently applied if the engine is performing within the given performance conditions. In fact this is very difficult to achieve, and hence the maintenance plan could be inaccurate. This research focuses on the variations of the remaining useful life of the turbine blade regarding the operating conditions, and consequently to the relevant losses that the operator could incur when maintenance actions and replacements are undertaken at less than optimum intervals, thus sacrificing the remaining useful life of the component. The method in use refer to the design lifing approach hence, the results taken could optimize the maintenance planning before the aircraft appears in service. However the same method could apply after the aircraft is in service and in combination with more accurate and detailed data to provide further optimization from the design lifing approach.

Take-off is a crucial stage of flight, and its affection on the engine's life will be investigated in the particular research. The recommended maintenance actions usually refer to when the aircraft is performing take-off with the engines at maximum thrust. However, these maintenance actions could be optimized further if the operator were able to constantly monitor the engine in combination with health management operations implementation, such reduced thrust at the take-off segment by exploiting a larger runway, or if the aircraft were to have a smaller gross weight. Before any results are presented, it would seem logical that the life of the engine would be seriously affected by the given thrust at the take-off segment as there is a direct link with the TET, and any increase in the TET increases the rate of creep failure.

For the present research, the maximum thrust of the engine refers to the calculated nominal value of achievable thrust provided by the manufacturer for the specific engine model, but is not concerned with throttle lever angles and other pure engine performance conditions. Therefore the reductions in thrust that will be performed later are based on the calculated reduced percentage of the given nominal maximum value. In order to investigate the reduced take-off thrust scenario in the present case study, an imaginary flight plan has been created.
According to this flight plan, an aircraft which is equipped with the engine detailed above constantly performs one particular route between city 1 and city 2. However, when it performs take-off from both cities it has the ability of reducing the thrust by taking advantage of the long runway. It could be expected that the loads on the engine would be lowered as the thrust reduces, and so the development of creep failure rate would also be lowered. Therefore, if the operator is to perform the health management operation of reducing the take-off thrust combined with condition monitoring of the engines for maintenance reasons, the benefits would be multiplied.

Specifically, the flight plan is divided into take-off, climb, cruise, descent and finally landing/reverse thrust. The first scenario that will be used in order to compare it with the rest will be as follows: take-off with 100% thrust, climb and descent at normal performance conditions, cruise at 30000ft and speed of 0, 8 Mach, and landing/reverse thrust with 0. 8 of the max thrust. The simulations and calculations for the climb and descent will be performed at the average altitude for both segments. For every segment of the initial plan there will be no deviation from the ISA conditions. The comparison will be achieved by changing just the take-off segment, while the others will remain constant for all the times and will be added after every different take-off condition in order to have a complete flight. The different take-off scenarios refer to 100%, 90%, 80%, and 75% of the max thrust. Furthermore, the previous thrusts will be matched for different ISA conditions at each time, which will be specifically -15, -10, -5, 0, 5, 10, 15, 22, 26, 32 ISA deviation. Finally, the last factor under investigation is the time the aircraft spends at take-off. Each of the above conditions will be matched for three different take-off durations of 1 min, 1.5 min, and 2 min. The following table 4.2 provides a general view of the concept that will be followed were the different each time take-off is matched with the other constant flight segments.

Table 4-2: indicative methodology of creation of the investigated flight profiles (Source: Author)

Case example 1. 100% take-off with ISA Dev. of -15	+ climb +	cruise +	descent +	landing
Case example 2. 100% take-off with ISA Dev. of -10	+ climb +	cruise +	descent +	landing
Case example 3. 100% take-off with ISA Dev. of -5	+ climb +	cruise +	descent +	landing
Case example 4. 100% take-off with ISA Dev. of 0	+ climb +	cruise +	descent +	landing

However, it is important here to define a number of crucial issues relating to the flight segments. Even though definitions of each are provided by the relevant authorities, it is very difficult in practice to isolate one flight segment from another. For example, as the relevant legislation indicates, the take-off segment ends when the distance from the ground is 35 feet, and after that the aircraft starts to climb. However the aircraft is still flying with the take-off thrust and it changes gradually to the maximum climb thrust after a specific time, rather than the distance shown above. In addition, sometimes some segments will be omitted in practice. An example of that would be a flight with a constant climb at a low rate and the opposite descent, omitting the segment of cruise. Therefore, for the present study the flight segments will be divided according the nominal thrust that is provided by the manufacturer for every flight segment. Therefore, essentially they are thrust segments that apply for the flight segments.

4.5 Thermal Model

As mentioned previously, creep growth is a result of the effects of temperature and stress on metallic components over the passage of time. The first thing that has to be done in order for the creep life to be calculated is the construction of the thermal model, which will demonstrate the condition of the blade in terms of the thermal forces and the temperature of the blade's metal. In the following section, only the process of calculations will be demonstrated, followed by an indicative example of one results table (Table 4.3). The completed results will be later discussed in the following chapter, along with graphical representations and a discussion of them. All the tables concerning the thermal model can be found in Appendix C.

Before any calculations are performed, it is essential to take into account a number of important assumptions for this research. Firstly, and most importantly, it is assumed that the methods used for blade cooling for the particular blade in question are convection and film cooling. These are the standard cooling methods for an engine such as the CFM-56 5B2, and the relevant calculations are achievable. Data relating to other cooling technologies used on this engine for the internal blade architecture, structure, material etc., are commercially confidential, and as such were unobtainable.

In addition, it is assumed that the effectiveness of the cooling methods remains constant, and this consequently leads to a fairly constant overall cooling effectiveness for every operational point of the engine. This is not completely accurate as the cooling effectiveness changes with the different operational points and it is mostly a matter of pressure ratios and bleed percentage (Han et al., 2000). Because of that deficiency the specific operational point must be set such that the effectiveness will be constant, and for the present study the conditions are those at flat rated take-off given at 30 ^{o}C.

The blade metal temperature is assumed to be uniform across the blade span, as is the distribution of temperature of the combustion gases at the annulus area of exit from the combustion chamber. Finally, it must be considered that many of the values are taken from graphs, and so a certain amount of inaccuracy is inevitable. The graphs needed are all provided in Appendix B. The margin of inaccuracy is compounded by the very small miscalculations resulting from the not totally real Turbomatch output data.

The procedure that has to be performed is simple and starts by using the following very simple equation of desired cooling effectiveness, where ε is the cooling effectiveness, Tb is the blade temperature, Tg the total gas temperature obtained from Turbomatch, and finally Tcin is the inlet coolant temperature, also obtained from Turbomatch.

$$\varepsilon = \frac{Tg - Tb}{Tg - Tcin} \qquad (4.1)$$

By using the above equation we are able to calculate the desired cooling effectiveness for the conceptual design of the chosen engine. However, because in this case some of the parameters of the equation are given from the simulations, eq. 4.1 reversed to eq.4.2, it is possible to calculate the temperature of the blade.

$$Tb = Tg - \varepsilon\,(Tg - Tcin) \qquad (4.2)$$

The cooling effectiveness from the above equation includes the overall cooling effectiveness, which in this case means the convection and film cooling effectiveness together. The calculation of them separately is based entirely on graphs provided in Appendix B. The first parameter that must be determined is the cooling flow φ. This can be achieved by referring to the graph A-2-1 which shows the cooling flow versus the stator outlet temperature (SOT). According to Walsh and Fletcher (1998) the SOT is 92%-93% of the TET, and knowing the SOT it is very easy to obtain the cooling flow value, albeit with a degree of inaccuracy because it was obtained from a graph. The next step is to find the m* coolant by using equation 4.3, where 2.5% is a typical value of bleed for modern turbine engines.

$$m^{*}\,coolant = \frac{\varphi\%}{2.5\%} \qquad (4.3)$$

After the m* coolant has been obtained, it is fairly easy to take the value for the corresponding convection and film cooling effectiveness by using the graph A-2-2. for m* cooling versus cooling effectiveness. The results can then be inserted into equation 4.4 to find the overall cooling effectiveness, after which they can be inserted into equation 4.2 in order to achieve the primary goal of finding the temperature of the blade (Rubini, 2009).

$$\varepsilon = \frac{\varepsilon_c + \varepsilon_f - \varepsilon_c\,\varepsilon_f\,(1 + \frac{1}{m\,coolant})}{1 - \frac{\varepsilon_c\,\varepsilon_f}{m\,coolant}} \qquad (4.4)$$

Table 4-3: Reference Thermal Model for Complete Flight (Source: Author)

Reference Thermal Model For Complete Flight with take-off at 0 ISA deviation

	ISA DEVIATION	TET (K)	SOT (K)	φ %	m*	ε_c	ε_f	ε (overall)	Tg (K)	Tcin (K)	Tb (K)	Tb (C)
Take-off	0	1504	1398.72	7.459	2.98	0.68	0.48	0.614	1504	837.05	1093.96	821.816
Climb	0	1496	1391.28	7.459	2.98	0.68	0.48	0.614	1496	835.54	1089.95	817.806
Cruise	0	1279	1189.47	7.459	2.98	0.68	0.48	0.614	1279	725.67	938.81	666.668
Descent	0	1159	1077.87	7.459	2.98	0.68	0.48	0.614	1159	667	856.52	584.373
Landing R/T	0	1413	1314.09	7.459	2.98	0.68	0.48	0.614	1413	792.75	1031.67	759.526

4.6 Stress Model

Monitoring in terms of life usage is not equally useful or useable for all internal engine parts. However, when we refer to critical life limited components, we mean primarily the rotating parts of the engine. The chosen component belongs to this category and the second step for the estimation of its remaining useful life via creep life analysis, is the creation of the stress model of the blade. Here as well only the process of calculations will be demonstrated, followed by an indicative example of one results table (Table 4.4). Again, the completed results will be presented and further discussed in the following chapter, along with graphical representations and discussion of them. All the tables concerning the thermal model can be found in Appendix D.

The value of the applied stresses is directly linked to the particular operational point at which the engine is working, in combination with the atmospheric conditions that prevail at the time. Of course, the temperature of the engine at the point at which such stresses are applied is a critical factor, and the fatigue and remaining life would be different for the same stress values but at different temperatures. The combination of temperature and stress values will provide the necessary data for the calculation of the remaining useful life of the component.

The sum of the stresses that are applied on a turbine blade refers to five major categories. Beginning with the weakest, there are complex stresses due to thermal gradients and shear stresses. These are followed by the bending stresses produced by the centrifugal load acting at a point which does not lie radially above the centre of the root datum section. Finally, the gas bending moments stresses, which can be subdivided into axial and tangential stresses result from the change of momentum and pressure of the fluid passing across the blade.

However, the majority of the applied stresses refer to the centrifugal stresses, and these account for 70-90% of the total stresses (Allison et al., 2010; Blackie et al., 2008). This kind of stress is linked to the rotational speed a(Allison et al., 2010; Blackie et al., 2008)nd the weight of the blade, and they can act in any section of the airfoil. As would be expected, this kind of stress is not equally distributed at all points of the blade. This is because every point on the blade is a different distance from the centre, and has a different inherent construction and shape, and so incurs different stresses. The literature reports studies that have been undertaken to identify the most vulnerable section for creep across the blade span (Eshati et al., 2010). In those studies the blade was divided into equal sections, which has not been done in the present research.

This is because in terms of maintenance and replacing there is no means of determining where the blade will fail first, as it will require replacement wherever that point is. This could be useful for studies that concern the design of new and stronger blades. Hence in the present study no divisions of the blade will be made, and the blade will be considered as a whole.

Furthermore, the division of the blade into sections would require more detailed data, and this is not available. At this point it is important to mention that it is assumed that the blade has a constant cross-sectional area, and any inaccuracies resulting from this are considered to be negligible. For a component of constant cross section as this is assumed to be, the calculation of the centrifugal stresses, which is the focus of this particular research, appears with the following equation

$$\sigma_{CF} = \rho * h * r_{cg} * \omega^2 \qquad (4.5)$$

Where:

σCF expresses the centrifugal forces

ρ is the density

h is the height

r_{cg} is the centre of gravity radius

ω is the angular velocity of the rotor

Table 4-4: Reference Stress Model for Complete Flight (Source: Author)

Reference Stress Model For Complete Flight with take-off at 0 ISA deviation										
	ISA Deviation	PCN	N (RPM)	ρ (kg/m^3)	h (m)	r_{cg}	chord (m)	ω	ω^2	σCF (Mpa)
Case 1	0	1.0403	15794.87	8700	0.0431	0.3451	0.0277	1653.98	2735671.83	354.307
Climb	0	1.0339	15697.70	8700	0.0431	0.3451	0.0277	1643.811	2702115.28	349.961
Cruise	0	0.9306	14129.30	8700	0.0431	0.3451	0.02731	1479.57	2189136.78	283.523
Descent	0	0.7958	12082.63	8700	0.0431	0.3451	0.0277	1265.25	1600864.86	207.334
Landing R/T	0	0.9693	14716.88	8700	0.0431	0.3451	0.0277	1541.10	2374997.89	307.595

4.7 Larson-Miller Parameter

The prediction of the remaining useful life is completed after the acquisition of the Larson Miller parameter. Essentially, this is a value that comes from experimental tests and represents the fracture or the rapture strength of the material in between the axis of applied stresses and the Larson-Miller parameter. Such graphs are available in the literature and they are unique for every type of material. In the following case the chosen material is the RENE N5. The master curve of the above particular material is shown in Figure 4.6 below.

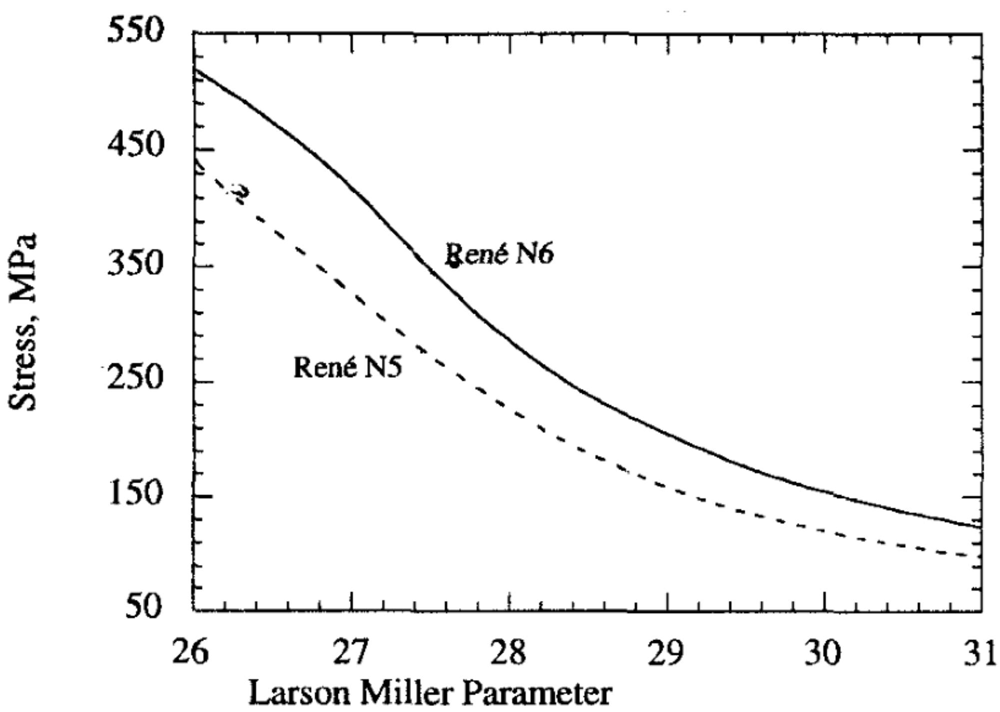

Figure 4-6: Rene N5 Larson-Miller Parameter Master Curve (Walston et al.)

In order to improve the accuracy of the results obtained, the above graph was plotted in Microsoft Excel. Nevertheless, there is still a degree of inaccuracy, but it is considered negligible. The plotted results are shown below in Figure 4.7.

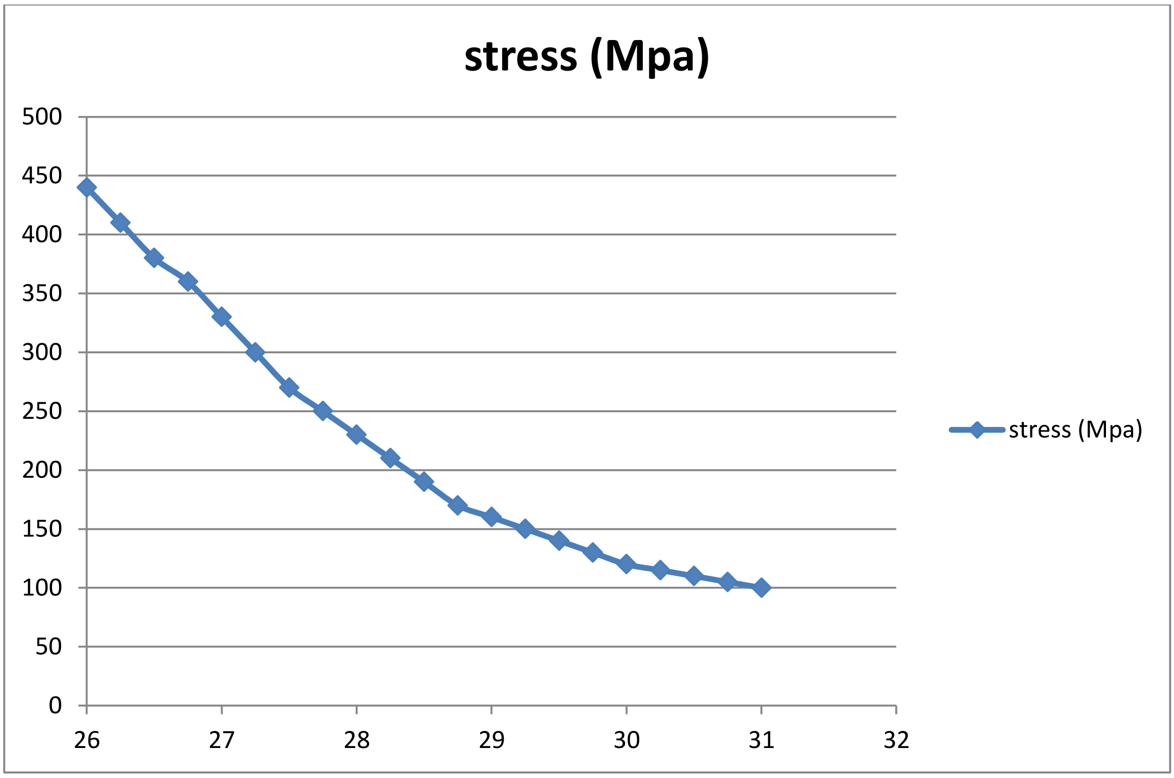

Figure 4-7: Rene N5 Larson-Miller Parameter Master Curve Plotted in Excel (Source: Author)

As it mentioned above, the Larson-Miller parameter is unique for every material, and it is expressed with the following equation, where LMP is the Larson-Miller parameter, T is the absolute temperature in K, tf is the time to failure counted in hours and C is a constant that takes value from 17 to 23. The value of C is determined by plotting log of time versus 1/ [T (°C) + 273] using rupture data from several tests at constant stress but different temperatures on the same material. For the present study the constant C is taken 20 as average an usual for aero gas turbine applications.

$$LMP = \frac{T}{1000}\left(log_{tf} + C\right) \qquad (4.6)$$

By looking at Table 4.7.1 above, it is obvious that the value of the Larson Miller parameter does not vary significantly for the calculated stresses obtained. In other words, because the stresses that were calculated in the previous section are from 290 MPa to 400 MPa the Larson-Miller parameter would vary from 26 to 28. Hence the accuracy in this measurement would be an asset. However because the values were obtained from the graph, it should be considered a certain level of inaccuracy. This, along with the miscalculations of the blade temperature, will lead to an inaccurate overall result, as it will be discussed later.

After some easy steps, equation 4.6 could be transformed for convenience to equation 4.9. Next, the process for the calculation of the remaining useful life can begin. It has to be noted that the calculated values for the remaining useful life refer to one targeted point, to which those conditions that were referred to as input apply. Hence this is not an indication of the real and total remaining useful life, and so it is essential to configure those results for a complete flight by cumulating them.

$$LMP = T\left(log_{tf} + C\right)10^{-3} \qquad (4.7)$$

$$log_{tf} = \frac{1000\,LMP}{T} - C \qquad (4.8)$$

$$tf = 10^{\left[\left(\frac{1000\,LMP}{T}\right)-C\right]} \qquad (4.9)$$

4.8 Cumulative Creep

A careful look at the previous calculations shows a very critical deficiency and inaccuracy, the result of which is data without any practical interest. More specifically, the previously calculated results concerning the remaining useful life of the turbine blade are sharing the lack of duration. This means that the calculations were done concerning only one time point at which the specific input conditions of stress and temperature were applied. However, despite the validity of the data, it is essential to implement the time duration in order to give a more realistic view. In any case, with the exception of temperature and loads, the time over these are applied is the paramount criterion for the occurrence of creep, and hence its consideration in the study is essential.

It is necessary first to specify the durations of the flight segments. With the exception of the cruise segment, which clearly depends on the distance of the journey, for the rest of the segments, reasonable values that correspond to real flights were used, without implying that they are standard for every flight (Zaretsky et al.). However, as mentioned previously, the take-off segment has three different values in order for their impact to be investigated. The table below shows these durations.

Table 4-5: Duration of the Flight Segments (Source: Author)

Flight Segment	Duration (min)	Duration (hours)
Take-off	1/1,5/2	0.0167/0.025/0.033
Climb	12	0.2
Cruise	150	2.5
Descent	12	0.2
Landing R/T	0.167	0.00277

However, it is crucial for the reader to accept that there is no deviation to the operating conditions of every segment. That means that for each of those, the atmospheric/surrounding conditions and the power settings remain constant, without their fluctuation affecting the study. However this is not the case in real conditions, with the most obvious occasions of climb and descent, for which atmospheric conditions are constantly and gradually changing throughout their duration. Therefore for the current study there will not consider these.

For each flight segment the real time to failure would be the true time spent in that particular flight segment, divided by the total calculated time to failure. It would also be possible for it the time to be counted in cycles, if this were felt to be more convenient, and if the duration of the cycle is determined initially; for example, one cycle is equal to two hours. However, even if it were possible to calculate with accuracy each flight segment as an individual entity, it would be difficult to calculate the time to failure concerning a complete flight by adding every individual flight segment. The requirement to determine the cumulative damage will be fulfilled by using Miner's law, which is essentially an inverse sum that accepts that the sum/total damage should be unity (equation 4.10).

In order for this to be achievable, Miner's law uses a linear damage sum assumption, relating the damage due to each flight segment to the ratio between time spent and time to failure. Additionally, the fact that it assumes that the fraction will occur when the sum reaches the unity is inaccurate. In reality, failure happens when the sum lies between 0.7 and 1.4. but this is something obtained in practice (Vigna Suria et al., 2006). Because of inaccuracies like this it is essential for the components to have a safety factor implemented when it comes to such analyses.

$$\sum_{i=1}(life\ fracture) = 1 \qquad (4.10)$$

$$\sum \frac{n}{N} = \frac{a}{N_a} + \frac{b}{N_b} + \frac{c}{N_c} + \cdots \qquad (4.11)$$

By using the above equations it is finally possible to obtain results with a practical meaning. A representative example is shown in below in Table 4.8.2. Further discussion about the sum of them is following in the next chapter along with graphical representation.

Table 4-6: Representative Result Table for Cumulative Creep (Source: Author)

Cumulative Creep for Complete Flight with Different Take-off thrust

take-off condition	t/tf for Take-off	t/tf for Climb	t/tf for Cruise	t/tf for Descent	t/tf for Landing R/T	Σ t/tf	R.U.L
take off 100% ISA dev 0	1.22701E-06	8.97597E-07	1.75499E-09	2.32042E-14	1.34065E-09	2.12771E-06	1379942.51
take off 90% ISA dev 0	1.45031E-07	8.97597E-07	1.75499E-09	2.32042E-14	1.34065E-09	1.04572E-06	2807731.619
take off 80% ISA dev 0	1.73747E-08	8.97597E-07	1.75499E-09	2.32042E-14	1.34065E-09	9.18067E-07	3198143.476
take off 75% ISA dev 0	5.69095E-09	8.97597E-07	1.75499E-09	2.32042E-14	1.34065E-09	9.06384E-07	3239369.32

Chapter 5: Results and Discussion

5.1 Introduction

This chapter discusses the results obtained through the research, and is the most important part of the study, as any data validation required will be presented here. As such, every effort will be made to present the discussion in such a way as to answer any questions that the reader may have.

The structure of this chapter will correspond to that of the previous chapter, and so every section will be autonomous and will be discussed separately, as far as possible. Initially, the chapter begins with the discussion concerning the accuracy and the reasonableness of the results obtained. This is because an effort has been made to provide the reader with a clear view of the importance and impact of the assumptions and miscalculations of the previous chapter, which could seriously jeopardize the primary aim of the research, namely the calculation of the remaining useful life of a turbine blade by using creep life analysis, and the impact that the take-off segment has on that life.

This chapter also includes the discussion of the results of the reduced take-off thrust health management operation, as this was selected to be the alternative in order to show and compare the different operating conditions and the impact of them on the take-off segment. The overall results, conclusions and achievements of the research are presented, with the term ‘overall’ indicating the advantages of the proposed diagnostic method, the contrast that is observed between the proposed diagnostic method and the methods of life estimation currently in use, and finally, the conclusions that the reader should arrive at the end of the thesis. Additionally, the financial benefits are outlined, along with suggestions for future work and future implementations.

It should be remembered that this research is submitted for the fulfillment of the degree of MSc by Research, for which there is a general requirement concerning the level of novelty that this research proposes. Thus, the novelty here is the estimation of the remaining useful life of a turbine blade as it relates to the primary failure mode of creep, and as it was observed for a particular model of engine, namely the CFM 56 5B2, which is not documented in the published literature.

5.2 Results of the Thermal Model

In general terms, the trend shown by the results concerning the thermal behaviour of the engine appears to be logical and as expected. However, the final overall judgement can be divided into two categories; accuracy of the results, and coherence and reasonableness. Specifically for this section, the data necessary for the calculation of the temperature of the blade, and thus the estimation of the RUL, were values of the turbine entry temperatures (TET) and cooling flow temperatures.

In terms of the first set of data (TET), it should be noted that there is no direct measurement from the engine as the harsh environment of this section does not allow for the installation of a sensor. Even if this were feasible, it would not be possible to obtain this data because they are protected by industrial confidentiality. Therefore, the Turbomatch simulation modelling was a necessity that inevitably resulted in a degree of inaccuracy in the results obtained.

Even if the fidelity of the tool does not concern the present research, an assessment of the reliability of the data provided is essential. An attempt at validation was made by comparing the TET values obtained for a given thrust, with data from other engines from different manufacturers, but with the same year of entry and technology, which were available in the literature. One such example was the V2522-A5, with a maximum thrust of 23041lbs. This is close to the 75% thrust scenario of this study, which has 23250lbs thrust. The TET obtained was 1458 K while that given by the manufacturer for the V2522-A5 was 1442 K. Given this comparison, it is the author’s belief that the data obtained by the simulation can be considered sufficiently accurate and reliable for the present study. Despite the fact that this comparison was obtained by means of confidential data, it is available for the reader to find other data in Roux (2007).

However, the greatest deficiency which negatively affects the accuracy of the results for the temperature of the blade arises from the values of the cooling effectiveness. This is affected by several internal and external engine factors, the most important being the bleed and the pressure ratios at each stage of the flight. This means that the cooling effectiveness does not remain constant throughout the flight, but instead is subjected to fluctuations. Furthermore, the inaccuracy is compounded by the fact that there are methods for cooling that were not considered in this research, and which are difficult to calculate because of the need for highly confidential data. An accurate estimation of this could be addressed in future research.

An example of the sensitivity of the final results of remaining life to small temperature changes is shown later in Section 5.4 (Table 5.3).This is because the temperature is basic term in the final equation (4.9) that calculates the remaining useful life, thus a small deviation produces an inaccurate result.

While these deficiencies inevitably affect the accurate calculation of the temperature of the blade, they are not harmful for initial and indicative studies like this one, and the results obtained by choosing to keep the cooling effectiveness constant proved to be perfectly adequate. This is because they are confirmed at a high level by another method found in the literature, the results of which are taken from a graph (Koff, 2004). According to this method, the overall cooling effectiveness can be found using graphs of the cooling effectiveness versus the TET.

On the other hand, despite the degree of inaccuracy in the actual values noted above, the results show coherence by being combined together into a graph and analysed. The graph that follows (Figure 5.1) shows the variation of the TET depending on the ISA Condition, with the former increasing as the ISA deviation increases. From any point of view this is absolutely logical. The greater ISA deviation indicates a greater ambient temperature. It is well known that the relationship between air temperature and air density is inversely proportional. That then has a direct effect on the operation of the engine, which requires greater effort to produce the same amount of thrust because the intake air density is reduced, and so more difficult to compress.

The effort made by the engine translates into greater fuel burn, and thus higher TET. This is confirmed for all of the thrust settings (individual lines in the graph). For example, when the engine performs with 100% thrust the TET increases for successively greater ambient temperatures (greater ISA deviation). As can be observed from the graphs, this increase is almost linear, even if the ISA deviations are not always equally divided (-15, -10, -5, 0, 5, 10, 15, 22, 26, 32). As expected, the trend that the TET follows is the same as that of the blade temperature during the ISA change (Figure 5.2). This is because the greater the TET, the greater the blade temperature.

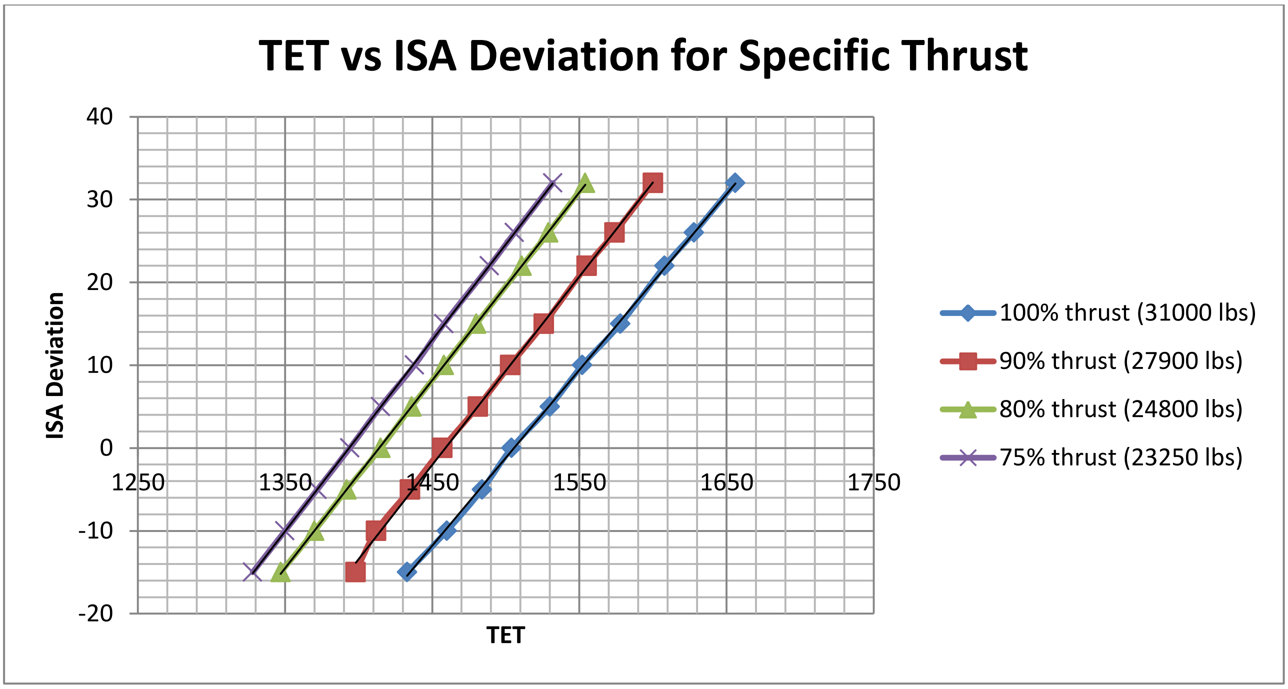

Figure 5-1: TET vs. ISA Deviation for Specific Thrust (Source: Author)

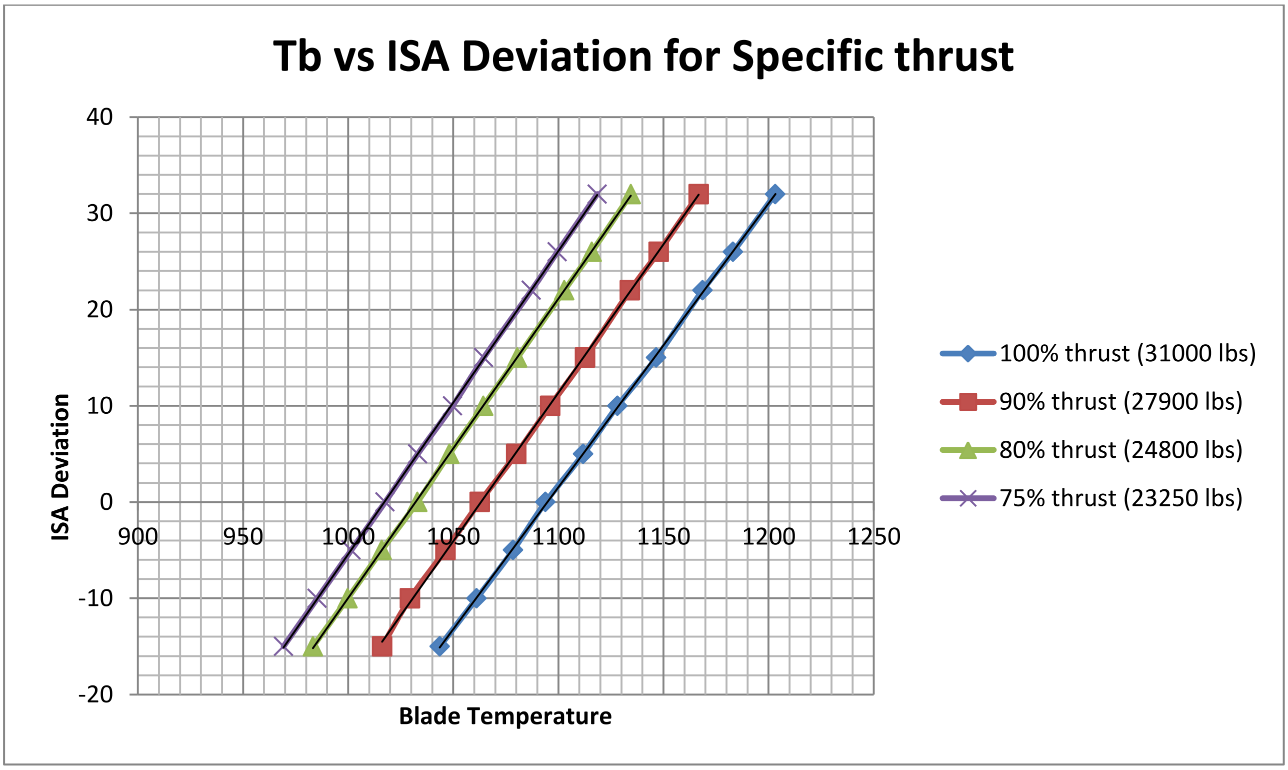

Figure 5-2: Tb vs. ISA Deviation for Specific Thrust (Source: Author)

The graph below shown in Figure 5.4 supplements the previous graphs. On this occasion the results are summarized in terms of specific ISA conditions (individual lines), rather than specific thrust. In other words, it shows the variation of TET for engines that operate in different thrust settings but under the same ISA conditions. Here as well there is a rise of the TET as the operator commands more thrust. This is also reasonable because greater thrust means greater fuel flow. The optimum conditions with the lower TET can be found at - 15 ISA deviation, which means an ambient temperature of 0 ^{o}C. However between optimum conditions (-15 deviation) and extremely harsh conditions (32 deviation) there is the middle measure of 15 deviation, which is also the flat rate temperature as proposed by the manufacturer, under which the engine is subjected to greater loads.

In any case, as the graphs indicate, as the ambient temperature drops there will be an equal drop in the TET because of higher air density, and hence lower effort. The issue of air density also concerns the climb and the descent of the aircraft, as the density gradually drops or increases as the altitude increases or decreases respectively. For the present study, those segments were calculated with average conditions, and hence the density that prevails at average altitudes. A good example to illustrate the effect of the air density would be to compare the general effect shown in Figure 5.3 with the more specific maximum take-off thrust of our engine, which is 31000 lbs at sea level, and compare it with the maximum cruise thrust, which is 5840 lbs at approximately 30000 ft.

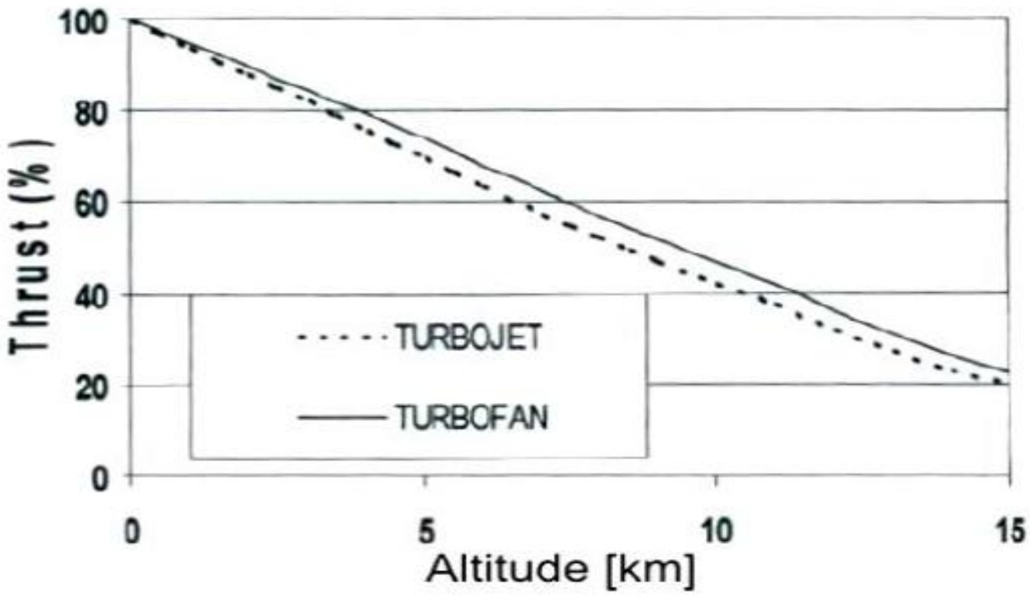

Figure 5-3: Effects of Altitude Density to Thrust (Bosdas et al., 2009)

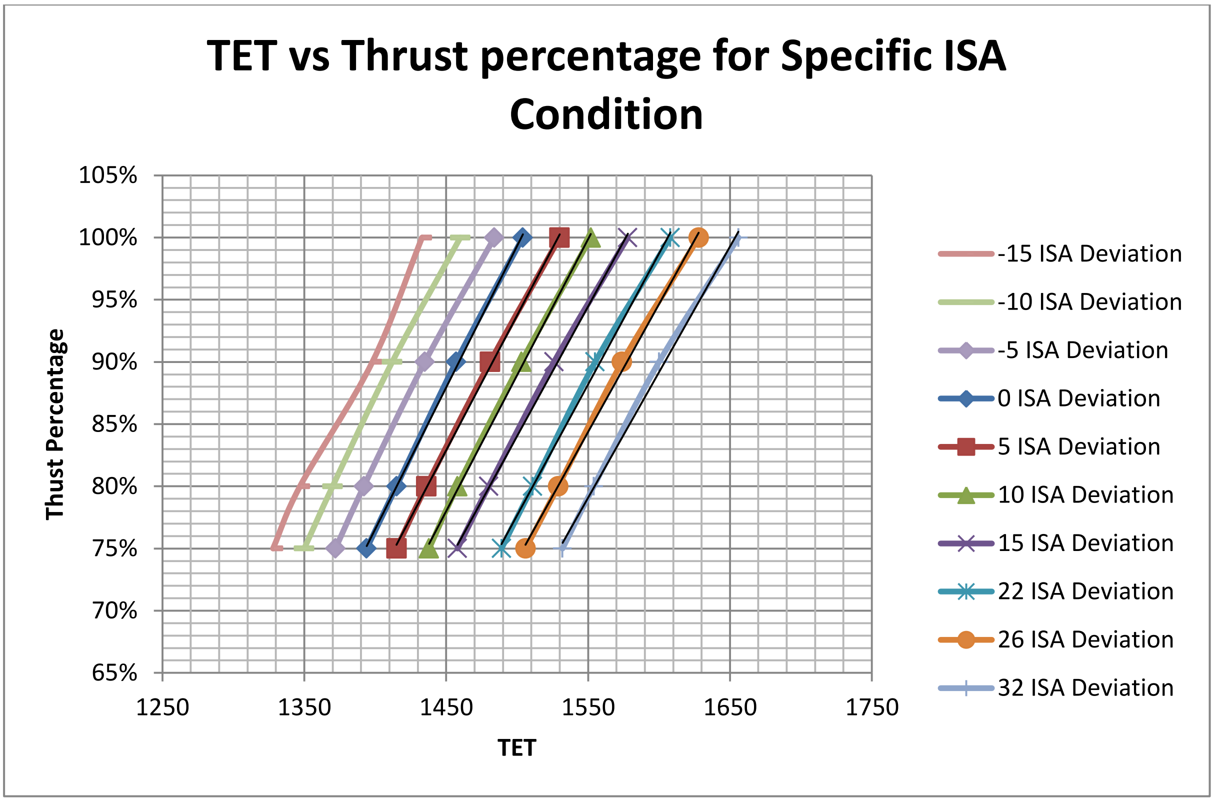

Figure 5-4: TET vs. Thrust Percentage for Specific ISA Condition (Source: Author)

5.3 Results of the Stress Model

Following the order of the previous chapter, the discussion at this point turns to the results of the stress model. Although it would be desirable, it is not possible to subdivide the discussion of these results into two categories, as was done for the results of the thermal model, because real values for the RPM, which are necessary for the construction of the stress model, were not available, and as such their correlation with those obtained from Turbomatch was not possible. Thus, only the reasonableness will be discussed here.

As noted earlier, as the ambient temperature increases, the engine requires more effort to produce the same amount of thrust, and it achieves this by increasing the fuel flow. However, the increased fuel flow is actually the price and not the effort, as the latter is described in terms of the increase in the RPM. The graph shown in Figure 5.5 below shows the RPM versus the TET for a specific thrust setting. The gradual rise of the TET shown in the graph is because every point indicates an increased ISA deviation. It is very clear that the greater the ambient temperature, the greater the TET, and thus the greater the RPM in order to achieve the same amount of thrust.

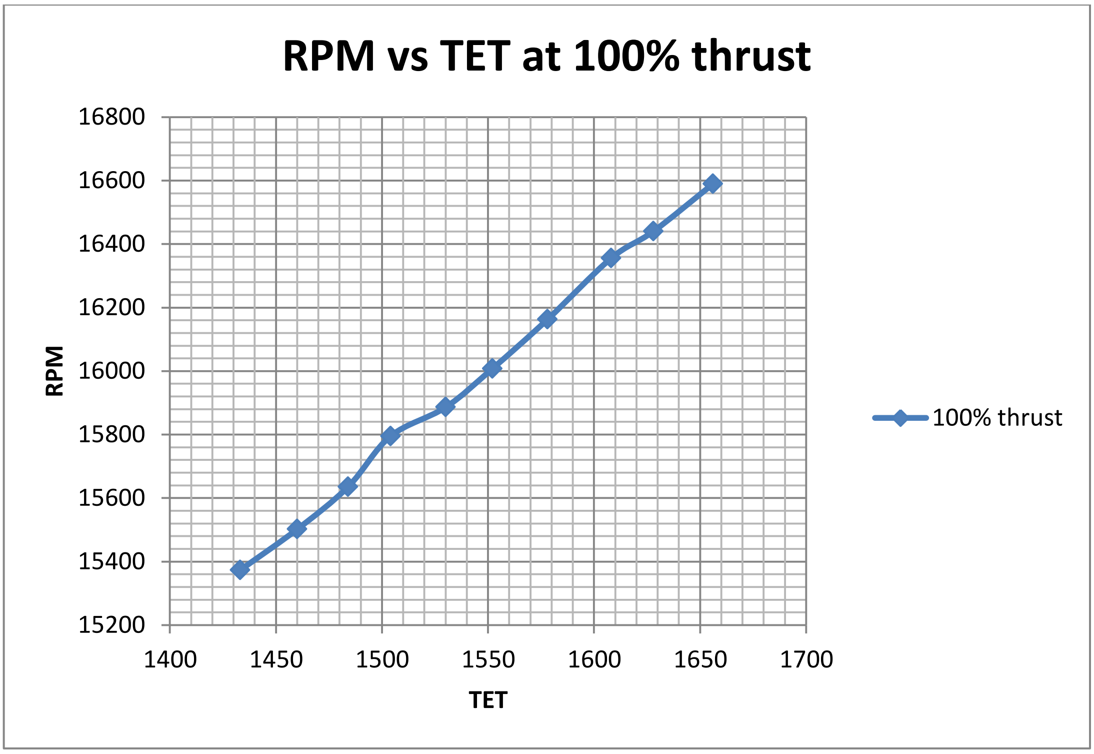

Figure 5-5: RPM vs. TET at 100% Thrust with Gradually Increased ISA Deviation (Source: Author)

It should be noted again that this research only addresses the centrifugal stresses, which account for the majority of stresses, and all that is necessary for their calculation is the RPM. However the majority does not also mean the sum. Hence, excluding a respectable 20%-30% of the total stresses each time has a successive impact on the LMP acquisition. As occurred previously for the thermal model regarding the temperature change of the blade, similarly a small deviation from the LMP, which is a basic term of the final equation (4.9) and is the outcome of the stress values provides totally inaccurate and misguided results. An indicative example concerning the Creep Life Results is provided in Table 5.2 in Section 5.4.

However, in many cases, the number of RPM seems to be greater than the maximum allowed, as shown in the stress calculation tables provided in Appendix D. Specifically, the value of PCN, which is the percentage of the maximum RPM, is more than 1 for almost all of the take-off scenarios.

This should not be considered a problem as a small percentage of tolerance is always built in by the manufacturer. Furthermore, the reading was taken when theoretically the engine was working at that given thrust each time (100%, 90%, 80%, and 75%). However, that level of thrust was not produced from the beginning of the take-off segment because the aircraft starts with zero velocity. Thus, even if the take-off segment lasts a very short amount of time in comparison with the other segments, the moment of maximum thrust is even lower, and the engine can afford to exceed the maximum RPM for such a short time.

Nevertheless, this does not mean that it is desirable to exceed the maximum permitted RPM, because the increase in the RPM would result in an increase in the centrifugal stresses. As can be seen in the following graph (Figure 5.6) the RPM follows the same trend as TET as the ambient temperature rises. The same increase is observed if the graph is reversed to show the variations as the thrust percentage increases for the same ISA condition (Figure 5.7).

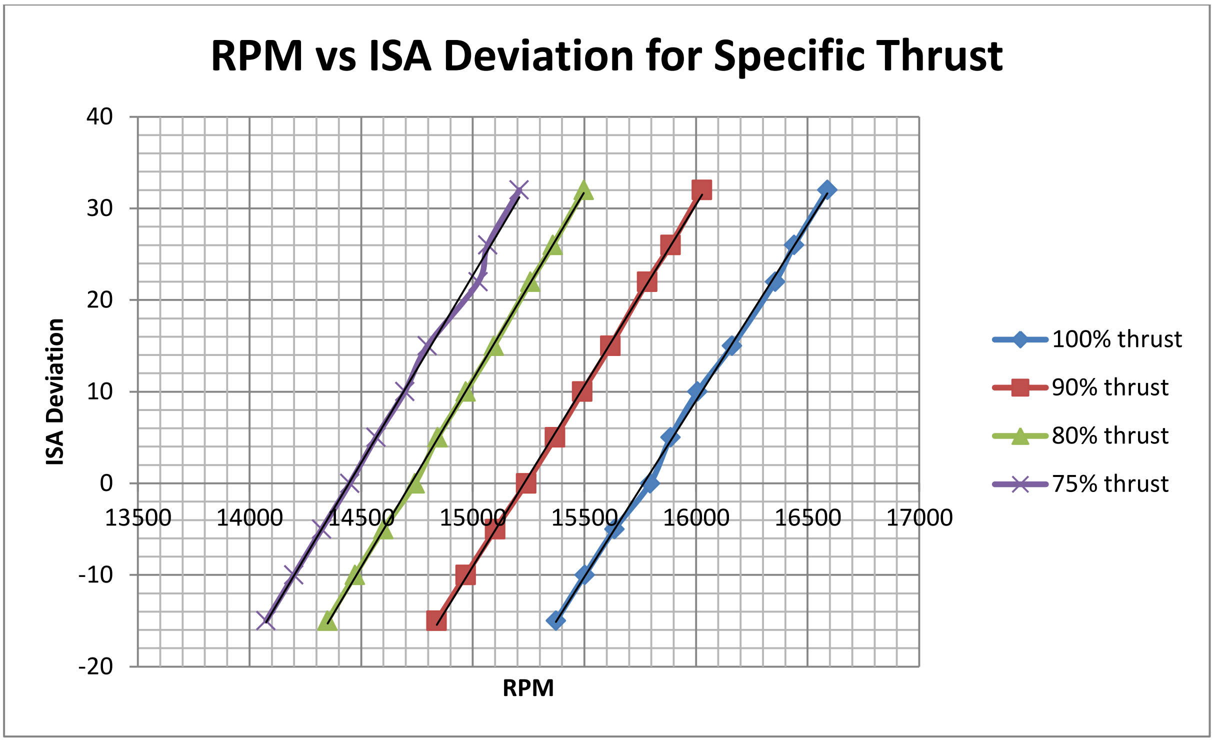

Figure 5-6: RPM vs. ISA Deviation for Specific Thrust (Source: Author)

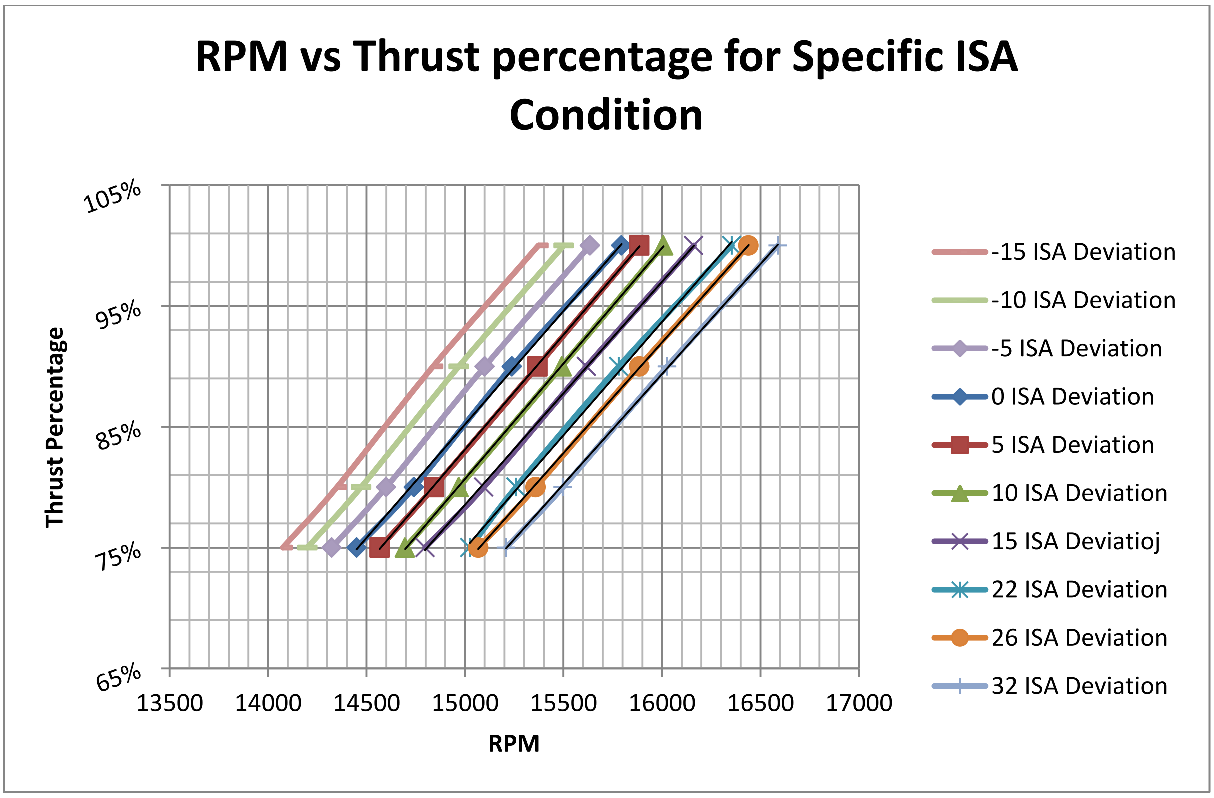

Figure 5-7: RPM vs. Thrust Percentage for Specific ISA Condition (Source: Author)

Because the source of the stresses is the RPM the same pattern of increase is followed by the stresses as the ambient temperature grows. This is clear in the following graph (Figure 5.8), which shows the stress rising for specific thrusts (individual lines), but with the ISA deviation increasing. As the graph in Figure 5.7 indicates, the same would happen if it were to show the stress rising for specific ISA deviation but with an increasing thrust.

However, the footnote of this section should be once more that for every gradual increase of ambient temperature or thrust, there is an equal increase in the centrifugal stresses that result from the increased RPM.

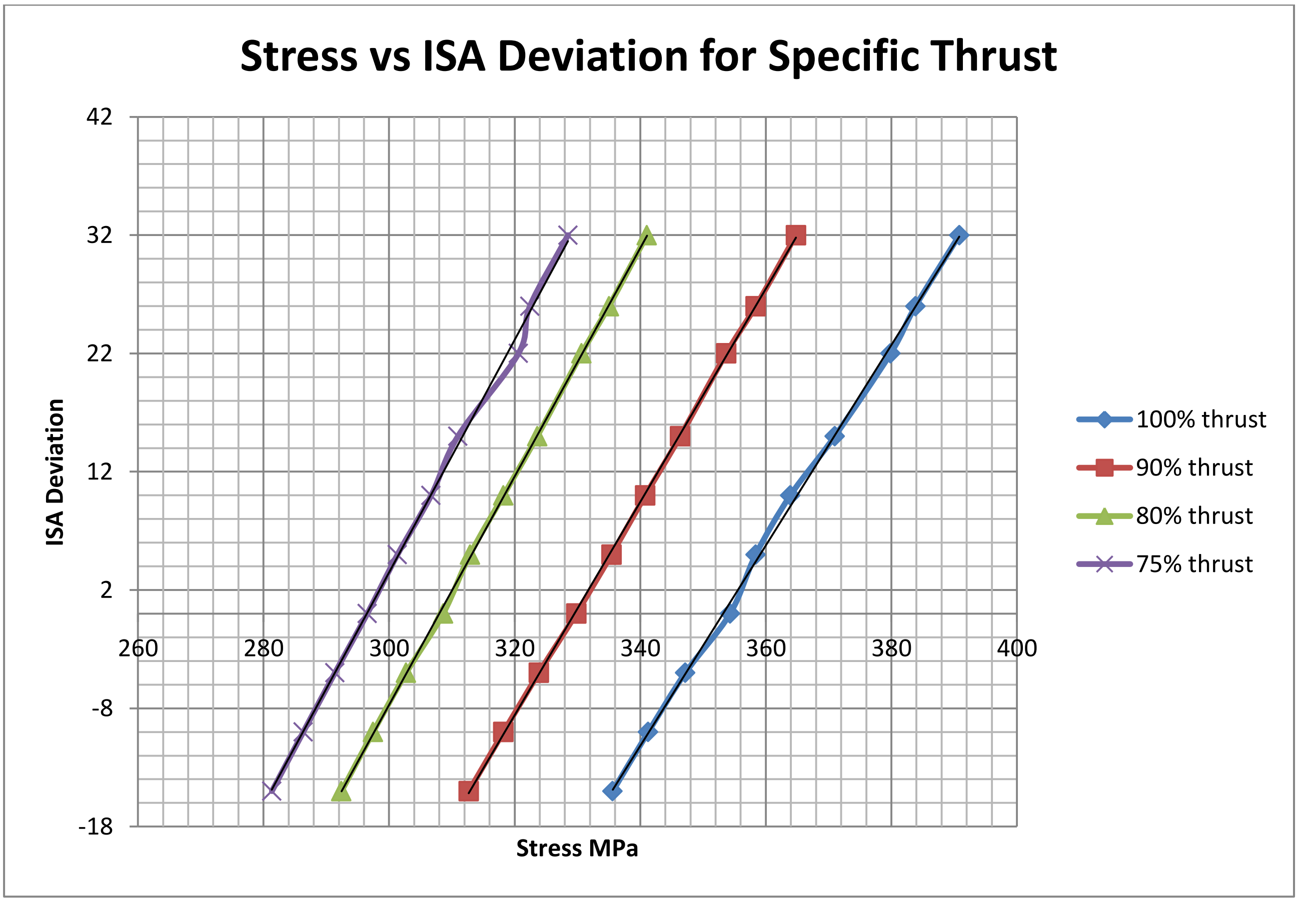

Figure 5-8: Stress vs. ISA Deviation for Specific Thrust (Source: Author)

5.4 Results of the Creep Life

As noted earlier, the rate of creep growth, and thus creep failure, is the result of raised temperatures and stresses on the component over time. First of all, it is essential to clarify that this research had arrived at certain results for the remaining useful life of the turbine blade as it concerns the primary failure mode of creep. However it is not clear how the accuracy of the results is affected as a result of the assumptions taken or from the presence of any miscalculation.

Beginning with the stress results from the previous section, it is essential to show how inaccurate calculations would be disastrous for the final result. The stress model was created because it enables the determination of the Larson-Miller parameter (LMP), which is the indicative value of the creep strength, and the values for the components that this research used range from 26 to 28. Even with such a small range, the smallest miscalculation of the LMP value could lead to huge deviations in the final results.

Unfortunately the LMP values had to be taken from graphs for this study, which is its greatest disadvantage, as is clear in the following tables, in which the resulting inaccuracies successively affect the cumulative creep calculations. Table 5.1 below shows the initial results for different take-off conditions and take-off duration of 2 min, calculated from the values of LMP obtained from the graphs, while Table 5.2 shows the value of LMP reduced by 0.1 at the author's discretion. It is clear that this 0.1 reduction in the LMP value comes with approximately 6000 hours reduction at remaining useful life of the blade.

Table 5-1: Initial Time to Failure Calculation (Source: Author)

		Creep Life Model Calculations			
Performance Condition	Calculated Stress	Calculated Tb	LMP	Calculated tf	t/tf
take off 100% ISA 0	354.30	1093.96	26.73	27166.24	1.22701E-06
take off 100% ISA 5	358.40	1111.89	26.7	10305.81	3.23442E-06
take off 100% ISA 10	363.90	1128.22	26.64	4096.65	8.13673E-06
take off 100% ISA 15	370.98	1146.65	26.57	1485.32	2.24418E-05
take off 100% ISA 22	379.88	1168.63	26.5	474.28	7.02814E-05
take off 100% ISA 26	383.84	1183.17	26.47	235.57	0.0001415
take off 100% ISA 32	390.82	1203.24	26.41	88.94	0.000374779

Table 5-2: Time to Failure calculation with change of -0.1 in LMP from Table 5.1 (Source: Author)

Creep Life Model Calculations					
Performance Condition	Calculated Stress	Calculated Tb	LMP	Calculated tf	t/tf
take off 100% ISA 0	354.30	1093.96	26.63	22009.95	1.51447E-06
take off 100% ISA 5	358.40	1111.89	26.6	8378.10	3.97862E-06
take off 100% ISA 10	363.90	1128.22	26.54	3340.36	9.97896E-06
take off 100% ISA 15	370.98	1146.65	26.47	1215.09	2.74327E-05
take off 100% ISA 22	379.88	1168.63	26.4	389.46	8.55876E-05
take off 100% ISA 26	383.84	1183.17	26.37	193.91	0.0001719
take off 100% ISA 32	390.82	1203.24	24.31	15.59	0.020848063

The inaccurate LMP value is not the only obstacle to accurately determining the remaining useful life. The other factor which could compromise the accuracy of the calculation is the temperature of the blade, calculated previously. Here too, the inaccuracies resulting from the assumptions made have a considerable impact on the accuracy of the results. This is clear in Table 5.3 where the calculations were made after a change in the temperature of the blade. One might imagine that 10 degrees more or less would not seriously affect the result. However, it can be seen that a rise in blade temperature of only 10 ^{o}C translates into a deviation of thousands of hours. Hence, all the assumptions made in order to complete the model have a significant detrimental effect on the final results.

Table 5-3: Time to Failure calculation with change of 10 ^{o}C from Table 5.1 (Source: Author)

Creep Life Model Calculations					
Performance Condition	Calculated Stress	Calculated Tb	LMP	Calculated tf	t/tf
take off 100% ISA 0	354.30	1103.96	26.73	16319.24	2.04258E-06
take off 100% ISA 5	358.40	1121.89	26.7	6295.66	5.29465E-06
take off 100% ISA 10	363.90	1138.22	26.64	2540.84	1.3119E-05
take off 100% ISA 15	370.98	1156.65	26.57	936.45	3.55953E-05
take off 100% ISA 22	379.88	1178.63	26.5	304.54	0.000109454
take off 100% ISA 26	383.84	1193.17	26.47	152.97	0.000217901
take off 100% ISA 32	390.82	1213.24	26.41	58.63	0.000568445

It is clear from the above that there are inaccuracies in the calculation of the remaining useful life of the blade. Nevertheless, in order to ascertain the degree of miscalculation, it is essential to provide realistic and accurate data concerning the service life of such a component. An attempt is made in the following tables (5.4, 5.5), which show the representative values concerning the service life of the components of an engine. The values may not be accurate for all the kinds of components, but it is a good guideline in order to provide a rough idea of the length of time that these components can be in service.

However, although these tables provide an indication of useful service life, it is still not accurate. This is because the components can hold out for much longer than that given value because a very generous safety factor is added, thus the given values below are after they have been multiplied by that. For example, as is indicated for a cargo aircraft, which is the closest available option to the example in this study, the total design service life is approximately 21500 hours. This estimation, however, also includes the safety factor, which for aviation applications is approximately around 0.6, so the initial calculated life is multiplied by that as a safety procedure.

Table 5-4: Design service life (USA DOD, 2004)

		Service Life			
System Category	Parts	Flight hours	Ground Run hours	Flight missions	Ground Runs missions
Fighter/Attack	Cold Parts	4,000	400	3,000	200
	Hot Parts	2,000	200	1,500	100
Bomber	Cold Parts	10,000	1,000	2,500	200
	Hot Parts	4,000	500	1,250	100
Cargo	Cold Parts	30,000	3,000	9,000	1,000
	Hot Parts	15,000	1,500	4,500	500
Trainer	Cold Parts	18,000	1,800	13,500	1,500
	Hot Parts	9,000	900	6,750	750
Helicopter	Cold Parts	6,000	TBD	3,000	TBD
	Hot Parts	6,000	TBD	3,000	TBD

Table 5-5 Design expected life (USA DOD, 2004)

	Design TACs	Life TAC*	Inspection TACs	Interval TAC*
Cold Parts (Durability Critical)	8,000	8,600	4,000	4,300
Cold Parts (Fracture Critical)	TBD	TBD	TBD	TBD
In-service Inspectable	8,000	8,600	4,000	4,300
In-service Non-inspectable	TBD	TBD	TBD	TBD
Deterministic	16,000	17,200	TBD	TBD
Probabilistic	8,000	8,600	TBD	TBD
Bearings (based on average bearing and engine installation tolerances)				
Ball (B1 value based on a material factor = 25)	4,000 EOH		4,000 EOH	4,300 EOH
Roller (B1 value based on a material factor = 6)	4,000 EOH		4,000 EOH	4,300 EOH

(NOTE: B1.0 life is the age at which 1.0% of the population will fail.)

	Design TACs	Life TAC*	Inspection TACs	Interval TAC*
Hot Parts (Durability Critical)	4,000	4,300		
Hot Parts (Fracture Critical)	4,000	4,300	4,000	4,300
Augmentor Module (excl. duct)	1,200	(total operation hr)	1,600	1,720
Ignitor Plugs High Energy	2,000	2,150	2,000	2,150
Low Energy	1,000	1,075	2,000	2,150
Augmentor	2,000	2,150	2,000	2,150
Expendable Items	4,000	4,300	4,000	4,300

* Includes ground operation and test cell and troubleshooting time.

In summary, it is important to clarify for the reader that the purpose of this section was to discuss issues around the technical part of the method, meaning the calculation process, and to address the impact of the assumptions made and any miscalculation of the fidelity of the final results. The overall conclusions about the method that refer to issues such as the advantages, the disadvantages, the practical meaning, and the possible future implementations are issues that concern the last section of this chapter namely 5.6. This is because even with the given percentage of miscalculation, the method has been reproduced for multiple operational scenarios in order for the remaining useful life of the blade to be calculated for all of them. Therefore, knowledge of the health management operations, and especially of reduced take-off thrust, and how every condition affects the turbine blade's service life was obtained, and it is the author's belief that this needs to be addressed before the final conclusions.

5.5 Conclusions for Health Management Operations

It should be noted that this particular research has gone beyond a simple reproduction of an already-established method for the calculation of the remaining useful life of a turbine blade. Since the study concerns the lifing approach performed at the design stage, it is necessary to highlight some optimum operational scenarios for the engine, and specifically to indicate the utilization of health management operations. Therefore the conclusions and the discussion that follow in this section refer to the significance and the impact of specific health management operations, which in this case is the reduced thrust take-off in conjunction with specific ambient factors. It should be noted here that despite the fact that the calculation of the remaining useful life has a certain level of systematic error resulting from the assumptions made, as explained in the previous section, the comparison and the correlation of them is possible as they all share the same degree of miscalculation due to the systematic error.

In order to perform health management operations it is essential to have a good knowledge of a wide range of operational scenarios in order to know which operation is the optimum one for the health of the engine by knowing the consequences from a wide range of conditions for the particular case. This could be translated into operations that optimize further the performance of the engine, the life of the engine, the maintenance cost for the engine etc. These optimization operations can be termed health management operations. For this particular research, the wide range of data obtained by means of different ambient conditions, along with the utilization of the reduced thrust take-off health operation, and its impact on the remaining useful life of the blade will be addressed.

Specifically, the research entailed the creation of 120 different take-offs by changing the ambient temperature, the altitude at which the take-off was supposed to be performed, the thrust setting, and the time spent. Every take-off condition was performed under different conditions each time, resulting in a different overall time to failure each time, after being added to the other flight segments as they were calculated previously for a single performance scenario.

Figure 5.9 is provided in order to investigate a stand-alone scenario for comparison, and to try to optimize it. As can be seen, for a specific and given amount of thrust, which in this case is the maximum of 100% of nominal thrust, the remaining useful life was calculated for a range of different ambient conditions (ISA deviations). Clearly, as the ambient temperature successively increases, the remaining useful life drops dramatically.

The explanation and the support for this comes from the results of the previous sections concerning temperatures and stresses. Since these are shown to be affected by the ambient temperature, and since these are the paramount values for the last remaining useful life calculation equation, these results should be negatively affected. Clearly, the greater the temperature, the lower the air density, hence more effort is required to produce the same results, leading to greater fatigue of the components, and hence the lower remaining life.

However the main thought here should be the fact that not all operators are “lucky” enough to operate in cold climates where greater the use of the engine before service is promoted, as shown in the same figure (Figure 5.9). An example of very hot climates might include Middle Eastern airports, providing the extreme ambient temperatures of 37 ^{o}C (22 dev.), 41 ^{o}C (26 dev.) and 47 ^{o}C (32 dev.). Therefore it is essential to focus on factors that will affect the process positively. One such a factor could be the time spent at the take-off section. The following Figure 5.10 shows the variations of the remaining useful life according to the time spent for the same thrust setting. It should be noted here that even if the take-off section is not as long, the engines are still performing with take-off thrust for approximately two minutes.

The observation shows that the results follow the same trend, but with a level of positive benefits in the remaining useful life as the time spent drops. However, the remarkable thing is that the benefits are not the same for all the ambient conditions even if the time held is reduced. In fact, above the 15 ISA deviation the saving in remaining time tends to be eliminated, which suggests that above this temperature, the time used for take-off does not play a significant role, hence another solution should be investigated. However, the lines also tend to cross for the opposite “cold” temperatures, which shows that under these conditions, the saving from the time reduction does not affect it greatly, and only the temperatures in the middle appear to be beneficial, albeit at a very low level.

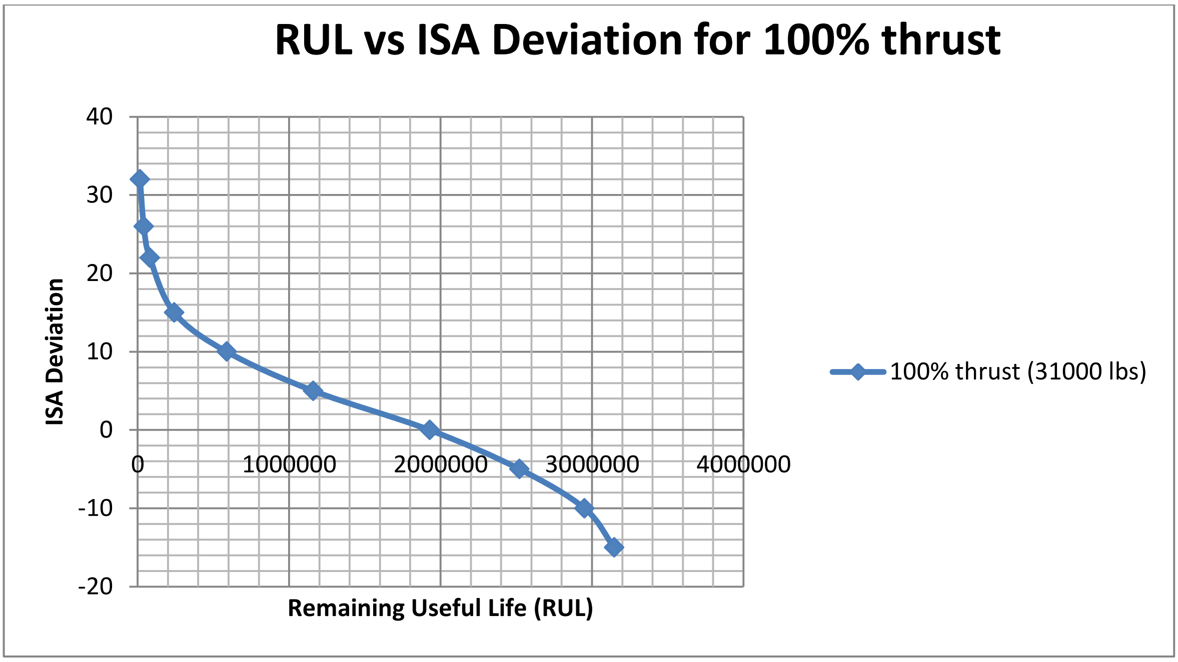

Figure 5-9: RUL vs. ISA Deviation for 100% thrust (Source: Author)

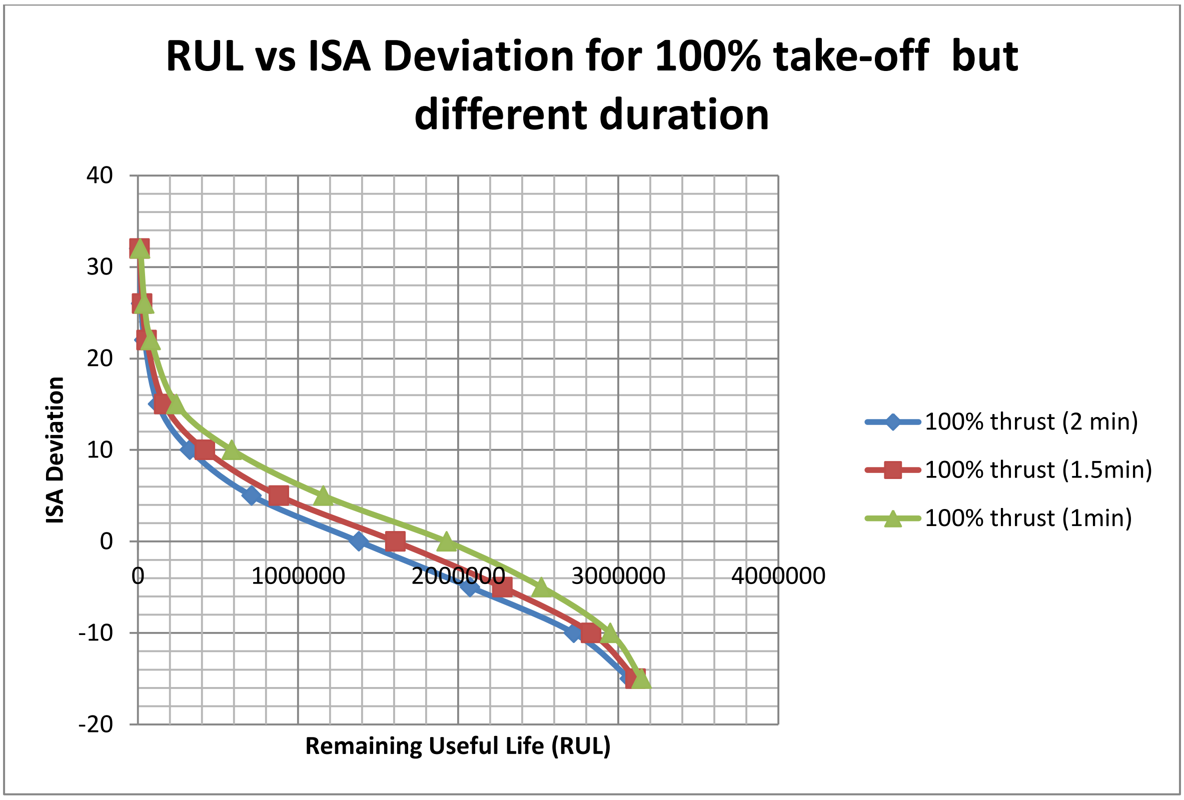

Figure 5-10: RUL vs. ISA Deviation for 100% take-off but different duration (Source: Author)

Therefore, because time reduction alone cannot be considered sufficient in order to optimize the remaining useful life of the blade, the next solution under investigation is reduced take-off thrust. This involves reducing the thrust in order to perform take-off for the same aircraft when certain circumstances, such as the payload of the aircraft, the available runway or the weather conditions allow it.

In the present case the remaining useful life was calculated for every operational point and also for every ambient condition under investigation. The expectations regarding the behaviour of the remaining useful life according to the changes made were confirmed and it would not bother to state from this point an initial verdict that reduced take-off thrust is working beneficial. Before the analysis of each graph, Figures 5.11, 5.12 and 5.13 are provided in order to give a plotted view of the results. Each line indicates a different thrust setting, while the points on it show the different ambient temperatures. Each figure indicates take-off durations of 1, 1.5 and 2 minutes respectively.

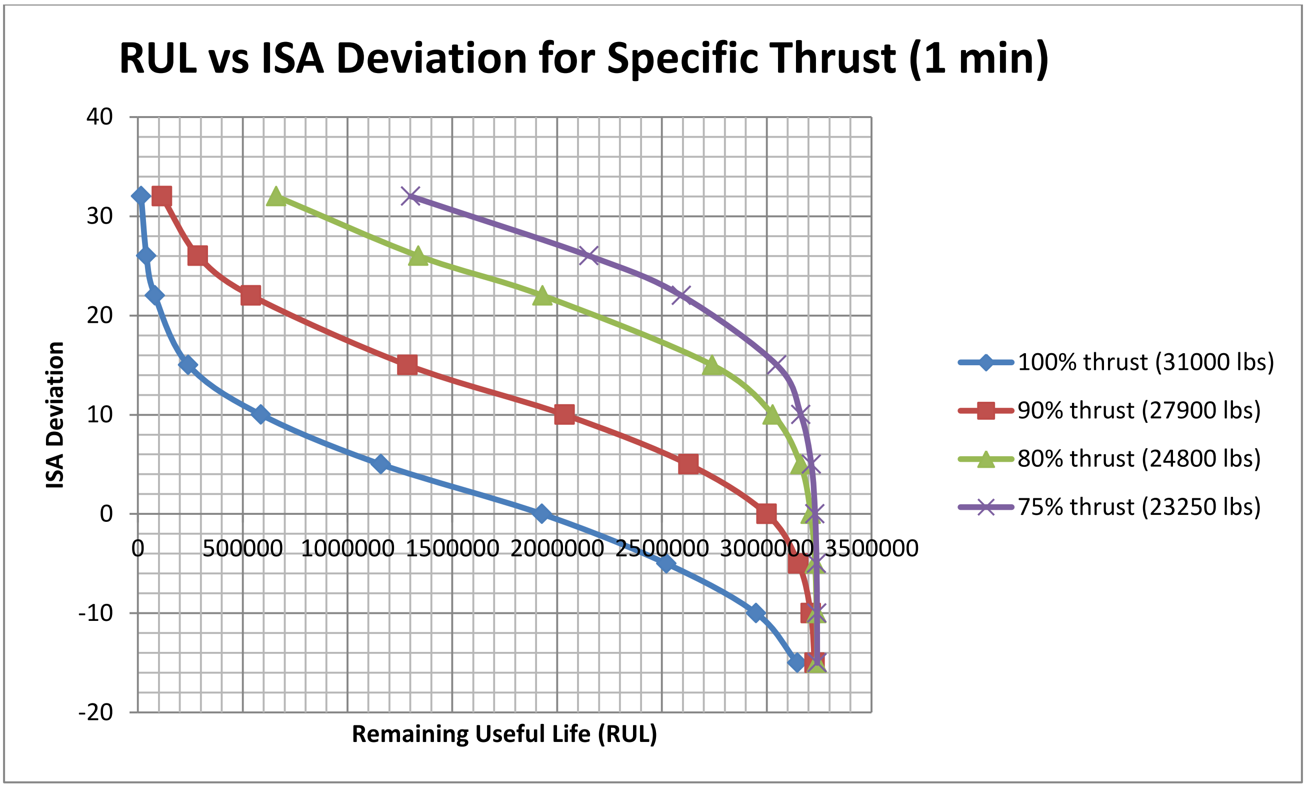

Figure 5-91: RUL vs. ISA Deviation for Specific Thrust (1 min) (Source: Author)

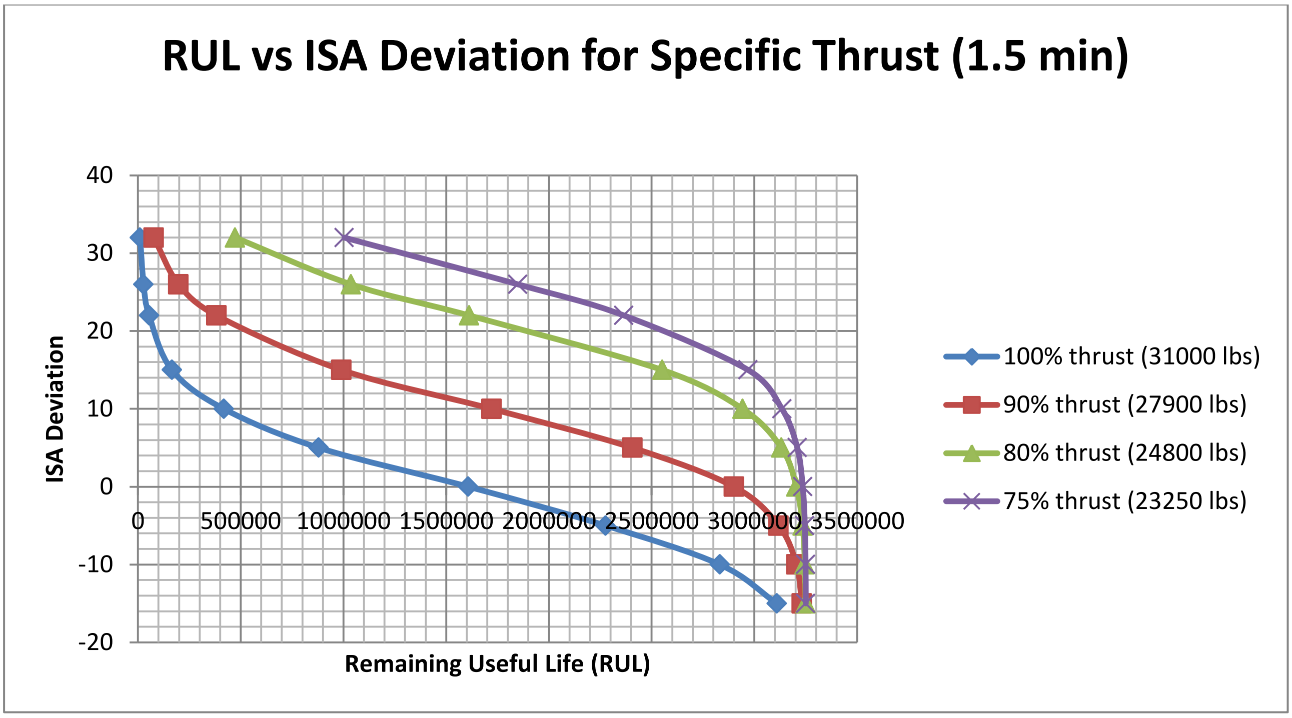

Figure 5-102: RUL vs. ISA Deviation for Specific Thrust (1.5 min) (Source: Author)

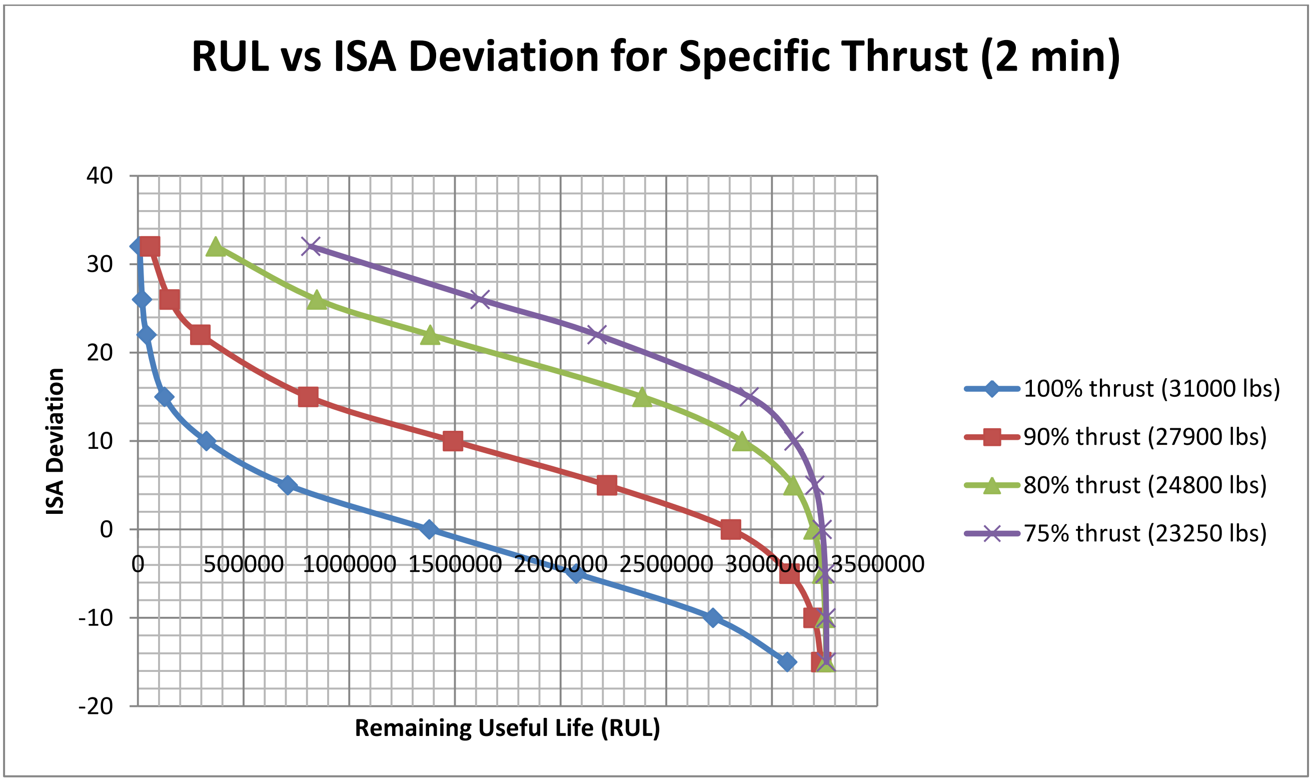

Figure 5-113: RUL vs. ISA Deviation for Specific Thrust (2 min) (Source: Author)

Clearly, for every different thrust setting an adequate change in the blade's remaining useful life can be observed. In other words, the variations in the amount of thrust combined with different ambient temperatures result in variations in the remaining life of the blade. In general terms it could be said that the lower the thrust, the greater the remaining useful life, although there are certain behaviours that require further discussion.

As previously noted, each line indicates a different thrust setting, and it is clear that as the ambient temperature falls the saving in the remaining useful life becomes greater. The same pattern applies for every different thrust setting, and the figure shows that for every successive lower thrust setting, the remaining useful life increases.

Nevertheless, in order to validate the reduced take-off thrust it is important to compare the operation for equal ambient conditions, as these are expressed by different spots on the individual thrust setting lines. To begin with, as is observed for the extreme ambient temperature of 32 dev., the remaining useful life increases as the thrust percentage drops. The most visible difference is from a thrust setting of 100% to 75% in which the hourly difference is of thousands of hours. However, the life savings are not always equal for every successive thrust drop. Specifically, for a reduction from 100% to 90% it is clear that there is no significant reduction (life saving), although that does not mean that there is no improvement. This indicates that both thrust settings, combined with the extreme ambient temperature, are already very "painful" for the engine, and thus there is no benefit.

Moving down from the highest temperature, it can be observed that the distances between the lines - and hence the thrust settings – become more equalized, with the largest lying between 90% and 80% of thrust, as these two settings better represent the two edges. This is because the 75% setting is very close to 80%, and hence there is no significant improvement will result, while the 90% and 100% thrust settings are already very painful. However, for every successive thrust drop there is an equal saving in terms of remaining useful life. Once again, the only exception appears for the highest thrust settings of 100% and 90%, where the intervals between the lines are steadily growing and the equalization is observed at the average of 15 dev.

This point of 15 ISA dev. is given by the manufacturer as the flat rated temperature at which the engine is subjected to greater loads. This specific temperature appears to be important because at this point every successive thrust reduction results in remarkable savings, with the greatest of them again appearing between a thrust reduction from 90% to 80%. It could also be stated that this point is a threshold below which an inverse situation is observed.

This means that as the ambient temperatures drops, the range of savings too becomes smaller. This can be seen by looking at each line that represents the different thrust settings. These lines tend to meet each other as they reach the lowest temperature point, that is the -15 ISA deviation, which in practice means an ambient temperature of 0 ^{0}C. It is important to remember that the distance between the lines is essentially the saving in remaining useful life. The savings in terms of remaining life tend to eliminated although that doesn't mean that there are none at all.

In practice, this means that for low temperatures the option of reducing the thrust in order to perform take-off does result in significant benefits in terms of remaining useful life. The benefits are very small, especially for the lowest temperature point, even if the reduction reaches 25% of the maximum. This is because the density of the air is at the highest levels and the fatigue is equalized.

These small savings could play a significant role in very extensive engine exploitation. Additionally, in such cases, factors like actual thrust needed and fuel burn should be considered in order to optimize the savings. It should be noted here that the choice to use 75% as the lower threshold under investigation was initially taken arbitrarily. In fact, as was found for the specific model, the manufacturer gives 75% as the lower threshold, thus the initial choice of the research is confirmed. For both high and low temperatures the same pattern is followed for all the take-offs, regardless of the duration.

The same conclusion can be arrived at from the following figures. However, in these cases the results are plotted differently to emphasise the different ambient temperature more clearly. Thus, Figure 5.14 shows the same results plotted for specific ISA conditions (individual lines) as the thrust percentage rises. Similarly, Figures 5.15 and 5.16 show the results for take-off durations of 1.5 and 2 minutes respectively.

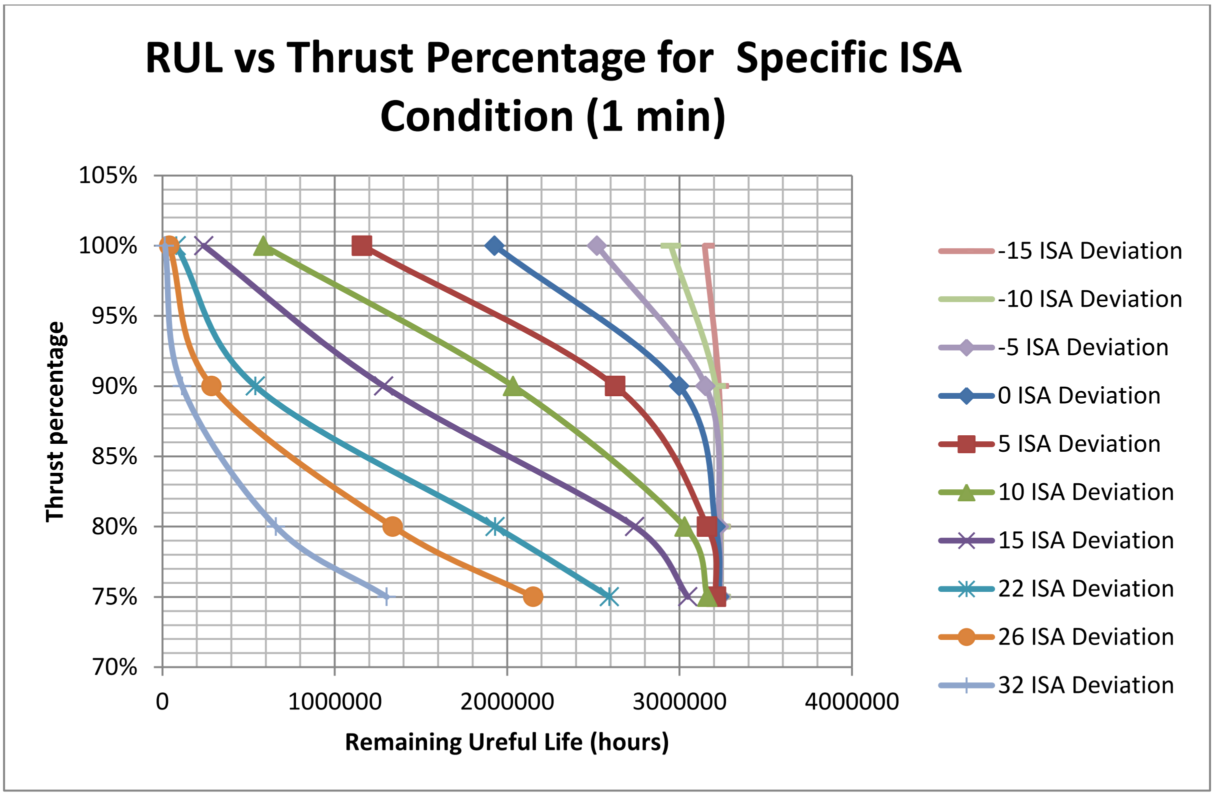

Figure 5-124: RUL vs. Thrust Percentage for Specific ISA Condition (Source: Author)

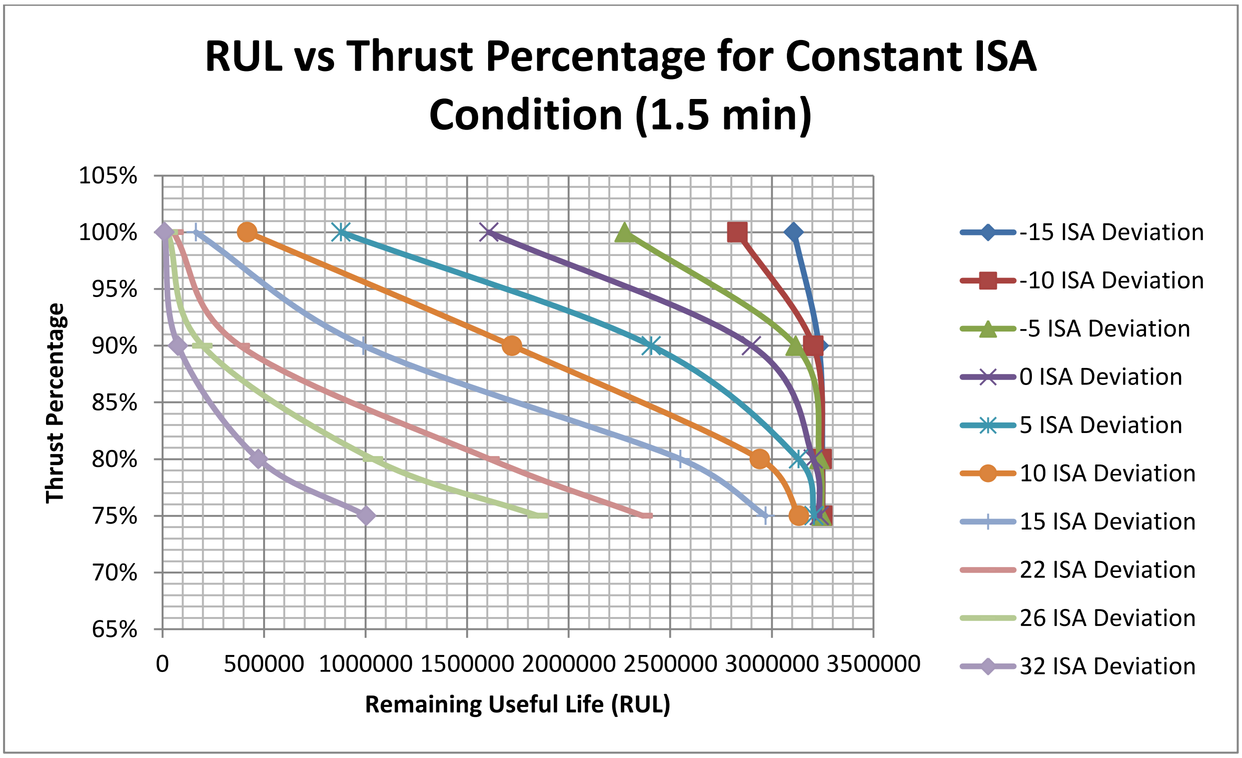

Figure 5-135: RUL vs. Thrust Percentage for Constant ISA Condition (1.5 min) (Source: Author)

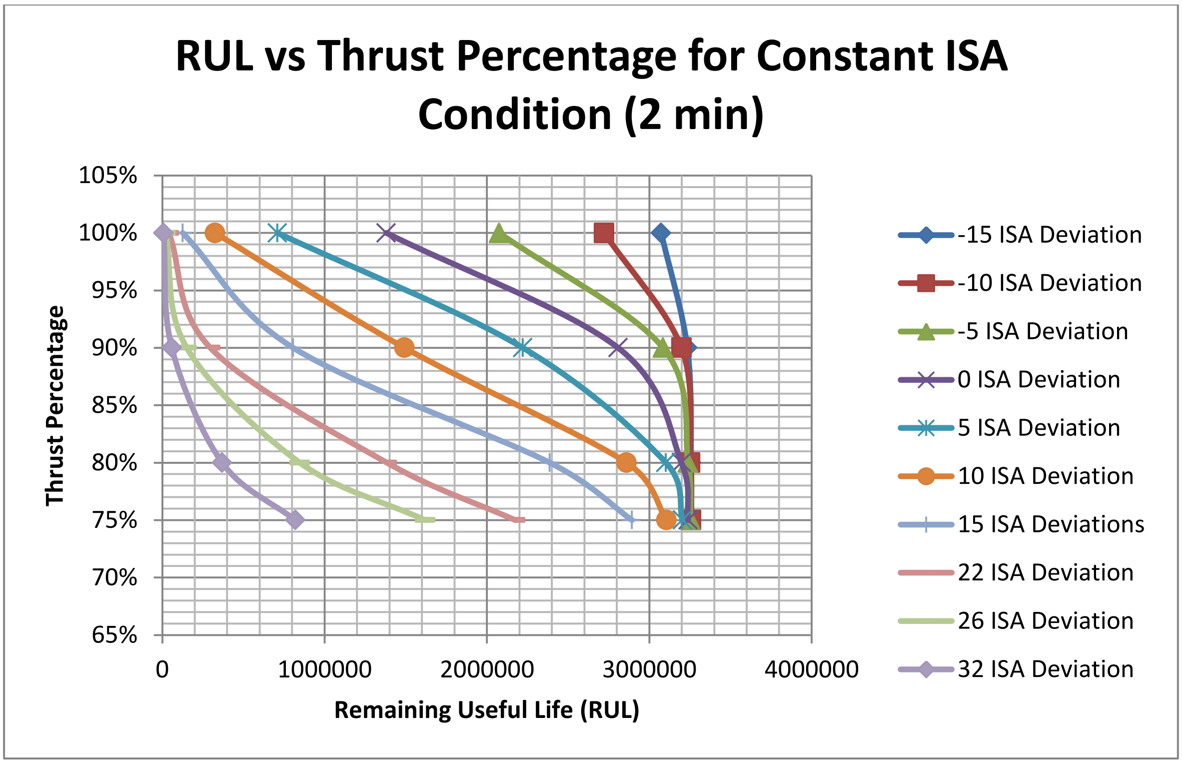

Figure 5-146: RUL vs. Thrust Percentage for Constant ISA Condition (2 min) (Source: Author)

In contrast to the previous graphs, in this case a different polarization is observed, which is expressed in terms of temperature, rather than thrust percentage. In other words, the individual lines which now express the different temperatures, and not the different thrust settings, tend to concentrate at the top left of the graph and at the bottom right.

Clearly, when the thrust setting remains high the drop in ambient temperature does not play a significant role. Fortunately, this does not apply for every ambient temperature and the savings are enhanced for middle temperatures. The reason why this was so obvious in the previous graphs is that for the lower limit of reduced thrust there is a threshold temperature of around 10 ISA dev. below which the life savings do not follow the same rapid pace of reduction as for the higher temperatures. Therefore, the main conclusion for the reduced take-off thrust health management operation is that the significance of the change is not constant and a balance between ambient temperature and thrust setting should be maintained. The overall conclusions are shown in the following sections.

5.6 Overall Conclusions

The following section presents a summary of this research, and the conclusions of its findings. It should be remembered that the purpose of this study was to conduct research on aircraft maintenance and to further investigate the concept of condition based monitoring maintenance by means of creep life analysis for turbine blades, in order to predict incipient failure and to identify when the next maintenance should be scheduled.

For many parts of the thesis conclusions should be expressed, but it is the author's belief that the starting point should be by answering the most important question: "is condition monitoring maintenance achievable?" The answer should be divided into two sub-sections, with the first answering the first part of the questions, which relates only to condition monitoring, without asking whether condition monitoring maintenance is achievable. As described in the second chapter, a number of monitoring (diagnostic) methods are in use in order to monitor of the condition of a component, in this case the aero engine. Because their ultimate goal is diagnosis, the monitoring element is essential, and therefore the ability of these techniques to produce assessments about the condition of the component is the key for the maintenance optimization process. This is because condition monitoring is the tool that reports on the condition of the component.

Having concluded that monitoring is in fact achievable, we move on to the second part of the question, and it is worth remembering that condition monitoring maintenance is essentially the actions taken after serious consideration based on approved diagnostic methods. This is also the ultimate answer that yes, condition monitoring maintenance is achievable because by utilizing condition monitoring techniques, we can achieve condition monitoring maintenance.

For this research the approved diagnostic by the use of which we can arrive at conclusions and make suggestions for the optimization of the maintenance process was creep life analysis by means of engine usage diagnostics. Despite the inaccuracies in the results, this research has managed to arrive at certain results which enable a discussion to take place. Assuming that the results have been perfectly calculated, this presents a clear indication of when the turbine blades need to be replaced, as it is expressed in terms of hours of remaining life.

By using the results of such an approach the operator is able to predict with a high degree of accuracy exactly when maintenance activity is required, resulting in optimum maintenance process scheduling. Being able to predict exactly when maintenance will be required is a primary requirement in aviation maintenance, and the utilisation of this approach offers a significant benefit in terms of predicting the need for maintenance activity.

Along with the high level of maintenance scheduling comes the avoidance of unwanted losses of the component‘s service life. Even if the operator were able to remove the blade at the exact maximum service time recommended by the manufacturer for that specific component, there is still a significant 0.6 safety factor which should not be neglected. A reduction of that factor by at least of 0.1 could account for thousands of extra service hours. Of course, as this entails trusting the method 100% and ignoring the safety factor, and as such could not be achieved, it is necessary to arrive at the optimum possible extended service life.

Along with the savings that could be derived from the optimisation of the maintenance scheduling and the accurate exploitation of the component’s life, an additional advantage that this method offers is the optimisation of resources. In this context the word resources could also include human resources, but most important resource refers to logistics. In other words, the use of this method eliminates the need to store spare parts which are used for replacements. It is well known that even if those parts are not installed in the engine they have a life limit which must be followed. Therefore if they are not used, or will only be used for a percentage of their total life, this would incur considerable losses which could be eliminated by using the method.

Moreover, one great advantage of this method that also applies as an enhancement of the improvement previously mentioned is the fact that the data needed in order to arrive at the necessary results are already measured on an aircraft. This was noted in the second chapter were it was confirmed that the two most well-known maintenance enhancement systems, namely CMS and ACMS, monitor the parameters needed for the results. In terms of improvement this is very important because the fundamental requirements of data collection have been covered. Therefore, the improvement in the data manipulation and/or processing should be easier. On the other hand, the fact that the data are already being monitored is advantageous because it enables the creation of systems dedicated to that function to predict only the failure mode of creep by means of engine usage diagnostics.

This could be relatively easy to create and it could also be necessary if we contrast this particular diagnostic method with the methods that are in use today in order to identify the fault. Unfortunately, the most promising solution for creep detection is the borescope inspection. The advantages of creep life analysis in contrast to borescope inspections are numerous, ranging from the amount of time that is saved in order to perform the check, to the personnel needed to perform the check, but most importantly, to the difficulty of each method in being accomplished. However, it should be noted that the proposed method actually predicts failure, in contrast to the borescope, which diagnoses by detecting the failure.

The predicting capabilities of the method offer another advantage in giving the opportunity to perform lifing assessments even at the design stage of the engine (blade); the only difference being that the data needed will be simulated and not real, as was the case in this research. By making use of this option, the manufacturer could be in a position to make more accurate maintenance recommendation, but more interestingly, to make recommendations based on the specific operating conditions, including the ambient and the performance conditions. In other words, the operator could benefit by creating a range of health management operations scenarios and could choose to operate the engine according to his needs and contexts. The greatest advantage is that he be aware of the engine's condition before he enters it into service, but having already done so, he also has the opportunity to optimize the results by using real data, as described previously.

For example, each individual operator can apply his own requirements regarding his operational profiles to create the maintenance schedule that would best suit his needs. Operators from Middle Eastern airlines could benefit from studies like this, and could perform reduced thrust take-offs, as this was the health management operation scenario under investigation in this research because of the most obvious benefit of the exploitation of the engine for a longer period of time without maintenance, thus increasing the time during which the aircraft would still be in service. If the results were sufficiently accurate, accurate predictions of financial savings could be given. For example, if the reduced thrust operation resulted in 200 or more hours, the number of additional journeys multiplied by the price per seat would provide an accurate financial value for the increased engine life, which in turn would give operators the incentive to maximize engine life by optimizing flying conditions.

It could be concluded here that reduced take-off thrust is beneficial for the life of the engine. Support for this suggestion comes from an official statement from CFM International concerning the benefits of reduced thrust on the CFM models. According to the company, reduced thrust operation results in lower take-off EGT, fewer operational events due to high EGT, lower fuel burn over the on-wing life of engine, decreased shop visits, and hence lower maintenance costs, and finally improved flight safety. Furthermore, according to the same source, every minute at take-off accounts for at least 45% of the engine's maintenance costs (CFM International, 2000). The requirements for performing the take-off with reduced take-off thrust would be a larger runway or lower gross weight, neither which is difficult to achieve.

However, on the other hand, even if the results from the creep life analysis are very tempting, they do have a few disadvantages. The most important of these is the fact that the method could only be applied to components that suffer from the failure mode of creep and this targets mostly the parts of the engine that are located in the combustion chamber and the back (combustion chamber, turbine blades, nozzle guide vanes). However, as was shown in Figures 4.1 and 4.2, the hot section of an engine is the one that suffers most from downtimes and failures respectively. An additional disadvantage of the method specifically, but also of the category to which the method belongs, is the fact that it refers only to gradually and progressively growing failure modes that will definitely occur. Hence, the methods lacks the ability to detect anything unexpected, along with production of real time fault indication.

Therefore other diagnostics that are capable of providing at least the type of failure inside the engine are useful in order for the ground crew to be prepared after the pilot's notification of the failure. In every case an ultimate diagnostic/prognostic should enclose a range of such methods in order to cover every type of component, and every aspect of failure, and to be able to provide indications at the optimum time needed.

To conclude, condition monitoring is the state-of-the-art concept of providing maintenance for all aircraft. The maintenance intervals are becoming more accurate, the logistic losses are becoming smaller, and the overall facilities and personnel organization are improving. Additionally, the method can be used as the requirements of a health management operations system is effectively an additional consultant that will constantly provide optimum suggestions concerning operations and decisions that need to be taken.

5.7 Suggestions for Future Work

At this point some recommendations for future work are essential. It is the author's desire for the present research to be continued with the use of more accurate data. This means that the most obvious proposal is for the next study to minimise as far as possible the assumptions taken in the present one. The most crucial parts for which improvement would be advantageous are the issues of cooling effectiveness assumptions and the calculation of the rest stresses, with the exception of the centrifugal stress.

Furthermore, some extra operational scenarios could be made concerning other flight segment parameters, such as ambient conditions and durations. Additionally, a more precise estimation of the climb and descent segments would result in more accurate and reliable results.

However, the most interesting part of any future work might be the implementation of the concept in software, in order to perform the calculations automatically. This could be essentially a software package that could control such data, put them in order, and then perform the mathematical calculations.

The ultimate development of the present research would be the development, if not the creation of two on-board intelligent systems. The first, which does not absolutely have to be on-board is one system dedicated to detect/predict the creep failure mode, which will function as described before. The second should be the improvement of the thrust control unit in such a way as to be already programmed with a wide range of operational scenarios. Then, according to the profile of every flight, the system should pick up the optimum choice of thrust setting as this will be already programmed.

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Appendix A

For the present research it was necessary to use Turbomatch simulation code. Because it is not clear to the reader how the code is working it should be given a very rough explanation in order to form a general view. To begin with, in order to simulate situation for an engine it is essential to have its encoded model. This encoded model should be created for such condition to allow the correlation with the real ones. In other words in order to validate the code the initial model is created for conditions that the data are already known. For the present that condition scenario was the MAX CLIMB.

The process begins by dividing the engine to its components and creating relationships between them. Below in the first column with letters is shown the engine divided in stages. The numbers right to the stages declare the data, information that each station needs in order for the model to be functional. An observation in the model provided indicates that numbers with their data being right of them with their explanation. In order for the reader to find it more easily we are talking about the column that starts with the number 1 and reaches till 98.

After, the completion of that, the code is providing the result which was initially fed to it. Till this point it is said that the code is working to design-point. However, after its completion the operator is able to modify certain parameters in order to investigate is affection. This is called running off-design and it is performed by indicating to the code the number of the stage that needs change and entering after a different parameter. This is shown right after the end of the previously mentioned column. It has to be noted that the following given model is only the input file that the code needs. The results that show the overall behavior of the engine and how is this counted, are enclosed in the output file that is generated from the code after a successful run. Unfortunately, due to confidentiality issues it is not possible to present such an example.

```
!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!
      CFM56-5B2
!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!!

HIGH BYPASS TURBO-FAN ENGINE (2 SPOOL)

DESIGN POINT :MAX CLIMB
ALT 10670 m
MACH 0.8
OPR 35.5
THRUST 28.6 kN

OTHER PARAMATERS OBTAINED BY MATCHING TAKE-OFF CONDITIONS

THRUST 137.9 kN
BPR 5.5
Mass Flow 433.6 kg/s

////
OD SI KE VA FP
-1
-1
INTAKE  S1-2         D1-4             R200
COMPRE  S2-3         D5-11            R210   V5  V6
PREMAS  S3,16,4      D12-15                  V12
ARITHY               D80-84
COMPRE  S4-5         D16-22           R230   V16
PREMAS  S5,17,6      D23-26
MIXEES  S16,17,18
DUCTER  S18-19       D27-30           R220
NOZCON  S19-20,1     D31              R290
COMPRE  S6-7         D32-38           R240   V32 V33
PREMAS  S7,22,8      D39-42
PREMAS  S8,21,9      D43-46
BURNER  S9-10        D47-49           R250
MIXEES  S10,21,11
TURBIN  S11-12       D50-57,240,97           V51
ARITHY               D85-91
TURBIN  S12-13       D58-65,260,98           V59
DUCTER  S13-14       D66-69
NOZCON  S14-15,1     D70              R280
PERFOR  S1,0,0       D71-74,280,200,250,290,0,0
CODEND
BRICK DATA////
!  INLET
1  10670.0      ! ALTITUDE
2   0.0         ! ISA DEVIATION
3   0.8         ! MACH NO
4   0.995         ! PRESSURE RECOVERY

!  FAN
5   0.85        ! SURGE MARGINE
6   1.0         ! ROTATIONAL SPEED,N1
7   1.7         ! FAN PRESSURE RATIO
8   0.895       ! EFFICIENCY
9   0.0         ! ERROR SELECTOR
10  1.0         ! MAP NUMBER
```

```
11  0.0          ! VS.V ANGLE

!  BYPASS-MAIN
12  0.839    ! LAMDA (BYPASS RATIO 5.2)
13  0.0          ! MASS FLOW LOSS
14  1.0          ! PRESSURE FACTOR
15  0.0          ! PRESSURE LOSS

!  BOOSTER
16  0.85         ! SURGE MARGIN
17  1.0          ! ROTATIONAL SPEED
18  1.8         ! PRESSURE RATIO
19  0.875        ! EFFICIENCY
20  1.0          ! ERROR SELECTOR
21  1.0          ! MAP NUMBER
22  0.0          ! VS.V ANGLE

!  AIR BLEED
23  0.01         ! LAMDA (1%)
24  0.0          ! MASS FLOW LOSS
25  1.0          ! PRESSURE FACTOR
26  0.0          ! PRESSURE LOSS

!  BYPASS-DUCT
27  0.0          ! REHEAT SELECTOR
28  0.01         ! PRESSURE LOSS 1%
29  0.0          ! REHEAT COMB.EFFICIENCY
30  1.E6         ! MAX REHEAT FUEL FLOW

!  FAN NOZZLE
31 -1.0          ! SWITHCH, AREA FIXED

!  HPC COMPRESSOR
32  0.85         ! SURGE MARGIN
33  1.0          ! ROTATIONAL SPEED
34  11.6    ! PRESSURE RATIO
35  0.875        ! EFFICIENCY
36  1.0          ! ERROR SELECTOR
37  3.0          ! MAP NUMBER
38  0.0          ! VS.V ANGLE

!  ACCESSORY BYPASS
39  0.01         ! BYPASS RATIO
40  0.0          ! MASS FLOW LOSS
41  1.0          ! PRESSURE FACTOR
42  0.0          ! PRESSURE LOSS

!  HPT COOLING BYPASS
43  0.167         ! BYPASS RATIO
44  0.0          ! MASS FLOW LOSS
45  1.0          ! PRESSURE FACTOR
46  0.0          ! PRESSURE LOSS

!  BURNER
47  0.06         ! PRESSURE LOSS
48  0.999        ! COMBUSTION EFFICIENCY
49  -1.0         ! FUEL FLOW
```

```
!  TURBINE-HPT
50  0.0          ! AUX.WORK
51  -1.0        ! REL NON-D MASS FLOW
52  -1.0         ! REL NON-D SPEED
53  0.91        ! EFFICIENCY
54 -1.0          ! REL ROT.SPEED(COMP TURB=-1)
55  3.0          ! COMP NO. FROM LOW END
56  5.0          ! TURBINE MAP
57 -1.0          ! POWER LAW

!  TURBINE-LPT
58  0.0          ! AUX.WORK
59  -1.0         ! REL NON-D MASS FLOW
60  -1.0        ! REL NON-D SPEED
61  0.92         ! EFFICIENCY
62 -1.0          ! REL ROT.SPEED
63  1.0          ! COMP NO. FROM LOW END
64  5.0          ! TURBINE MAP
65 -1.0          ! POWER LAW

!  CORE DUCT
66  0.0          ! REHEAT SELECTOR
67  0.0          ! PRESSURE LOSS 1%
68  0.0          ! REHEAT COMB.EFFICIENCY
69  1.E6         ! MAX REHEAT FUEL FLOW

!  CONVERGENT NOZZLE
70 -1.0          ! SWITHCH, AREA FIXED

!  PERFORMANCE
71 -1.0          ! POWER (-1=TURBOJET/FAN)
72 -1.0          ! PROPELLER EFFICIENCY(")
73  0.0          ! SCALING INDEX
74  0.0          ! REQ'D D.P. THRUST

!  ARITHY : BOOSTER SPEED = FAN SPEED
80  5.0          ! COPY
81 -1.0
82 17.0          ! BOOSTER SPEED
83 -1.0
84  6.0          ! FAN SPEED

!  ARITHY : LPT WORK = BOOSTER WORK + FAN WORK
85  1.0          ! ADD
86 -1.0
87  260.0        ! LPT WORK
88 -1.0
89  210.0        ! FAN WORK
90 -1.0
91  230.0        ! BOOSTER WORK
97  0.0
98  0.0


-1
1 2  177.0          ! INLET MASS FLOW
```

```
10 6 1460.0       ! TET
-1
1  0.0     ! ALTITUDE
2  -15.0          ! ISA DEVIATION
3  0.0          ! MACH NO
-1
10 6 1548.0       ! TET 137.9kN 100%
-1
-1
10 6 1499.5       ! TET 124.1kN 90%
-1
-1
10 6 1455.0       ! TET 110.31kN 80%
-1
-1
10 6 1434.0       ! TET 103.42kN 75%
-1
1  0.0     ! ALTITUDE
2  -10.0          ! ISA DEVIATION
3  0.0          ! MACH NO
-1
10 6 1577.6       ! TET 137.9kN 100%
-1
-1
10 6 1524.2      ! TET 124.1kN 90%
-1
-1
10 6 1478.8       ! TET 110.31kN 80%
-1
-1
10 6 1457.6       ! TET 103.42kN 75%
-1
1   0.0     ! ALTITUDE
2  -5.0          ! ISA DEVIATION
3  0.0          ! MACH NO
-1
10 6 1603.0       ! TET 137.9kN 100%
-1
-1
10 6 1549.0       ! TET 124.1kN 90%
-1
-1
10 6 1502.8       ! TET 110.31kN 80%
-1
-1
10 6 1481.3       ! TET 103.42kN 75%
-1
1  0.0     ! ALTITUDE
2  0.0          ! ISA DEVIATION
3  0.0          ! MACH NO
-1
10 6 1624.0       ! TET 137.9kN 100%
-1
-1
10 6 1573.0       ! TET 124.1kN 90%
-1
-1
```

```
10 6 1528.0      ! TET 110.31kN 80%
-1
-1
10 6 1505.0      ! TET 103.42kN 75%
-1
1  0.0     ! ALTITUDE
2  5.0         ! ISA DEVIATION
3  0.0         ! MACH NO
-1
10 6 1652.0      ! TET 137.9kN 100%
-1
-1
10 6 1599.0      ! TET 124.1kN 90%
-1
-1
10 6 1550.0      ! TET 110.31kN 80%
-1
-1
10 6 1528.0      ! TET 103.42kN 75%
-1
1  0.0     ! ALTITUDE
2  10.0         ! ISA DEVIATION
3  0.0         ! MACH NO
-1
10 6 1676.0      ! TET 137.9kN 100%
-1
-1
10 6 1623.0      ! TET 124.1kN 90%
-1
-1
10 6 1574.0      ! TET 110.31kN 80%
-1
-1
10 6 1552.0      ! TET 103.42kN 75%
-1
1  0.0     ! ALTITUDE
2  15.0         ! ISA DEVIATION
3  0.0         ! MACH NO
-1
10 6 1703.0      ! TET 137.9kN 100%
-1
-1
10 6 1647.0      ! TET 124.1kN 90%
-1
-1
10 6 1598.0      ! TET 110.31kN 80%
-1
-1
10 6 1574.0      ! TET 103.42kN 75%
-1
1  0.0     ! ALTITUDE
2  22.0         ! ISA DEVIATION
3  0.0         ! MACH NO
-1
10 6 1736.4      ! TET 137.9kN 100%
-1
-1
```

```
10 6 1678.5      ! TET 124.1kN 90%
-1
-1
10 6 1630.5      ! TET 110.31kN 80%
-1
-1
10 6 1607.5      ! TET 103.42kN 75%
-1
1  0.0     ! ALTITUDE
2  26.0         ! ISA DEVIATION
3  0.0         ! MACH NO
-1
10 6 1757.5      ! TET 137.9kN 100%
-1
-1
10 6 1698.2      ! TET 124.1kN 90%
-1
-1
10 6 1649.6      ! TET 110.31kN 80%
-1
-1
10 6 1625.5      ! TET 103.42kN 75%
-1
1  0.0     ! ALTITUDE
2  32.0         ! ISA DEVIATION
3  0.0         ! MACH NO
-1
10 6 1787.3      ! TET 137.9kN 100%
-1
-1
10 6 1726.5      ! TET 124.1kN 90%
-1
-1
10 6 1676.9      ! TET 110.31kN 80%
-1
-1
10 6 1653.2      ! TET 103.42kN 75%
-1
-3
```

Appendix B

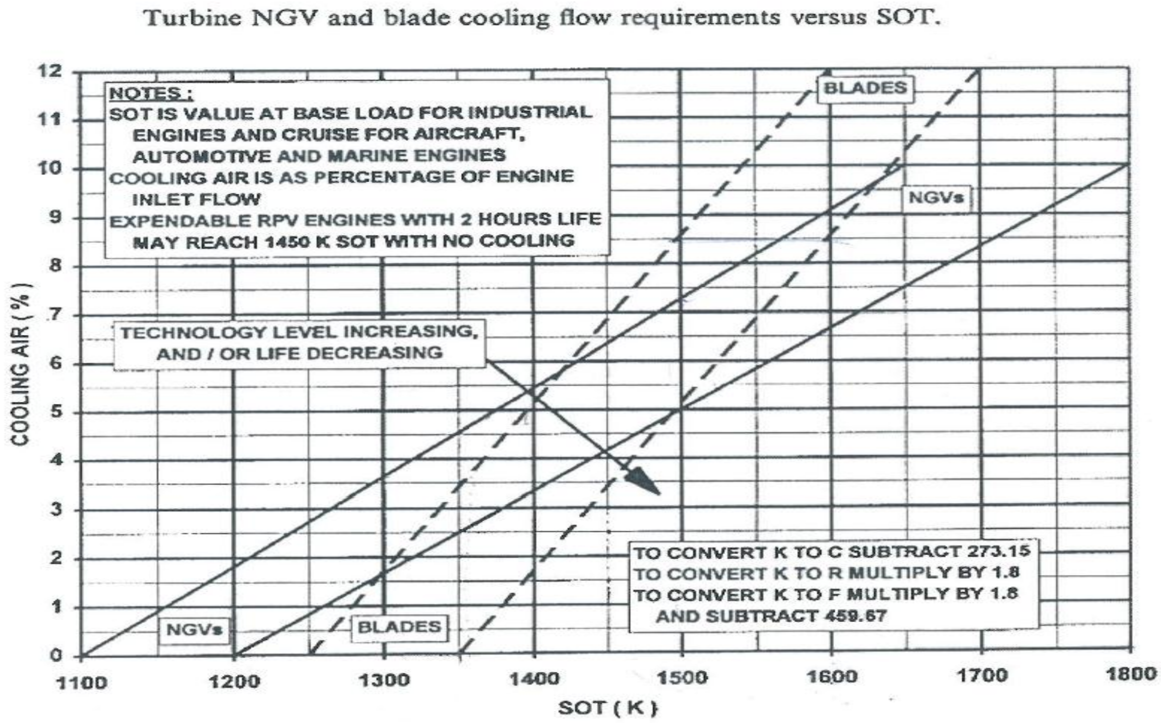

Figure A-1: cooling air vs. SOT graph

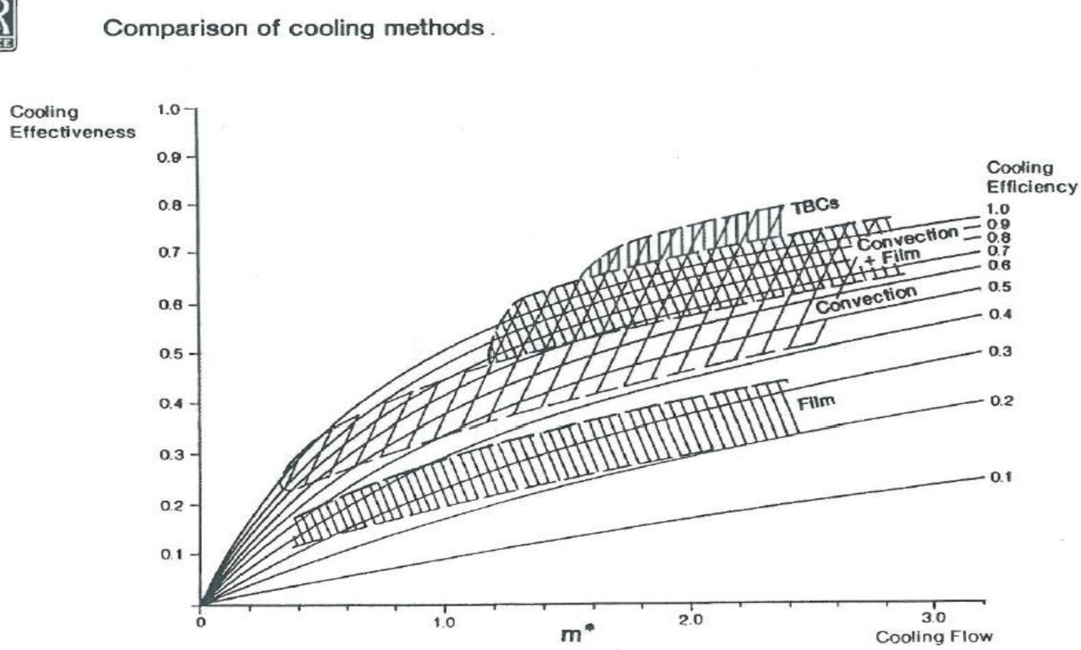

Figure A-2 cooling effectiveness vs. m* graph

Appendix C

Thermal model for Take-off at maximum thrust 31000 (lbs) / 137,9 (KN)												
	ISA DEVIATION	TET (K)	SOT (K)	φ %	m*	ε_c	ε_f	ε (overall)	Tg (K)	T cin (K)	Tb (K)	Tb (^{o}C)
case 1	-15	1433	1332.69	7.458	2.983	0.68	0.48	0.615	1433	799.61	1043.59	771.44
case 2	-10	1460	1357.8	7.458	2.983	0.68	0.48	0.615	1460	811.35	1061.21	789.06
case 3	-5	1484	1380.12	7.458	2.983	0.68	0.48	0.615	1484	824.45	1078.51	806.36
case 4	0	1504	1398.72	7.458	2.983	0.68	0.48	0.615	1504	837.05	1093.96	821.81
case 5	5	1530	1422.9	7.458	2.983	0.68	0.48	0.615	1530	849.92	1111.89	839.74
case 6	10	1552	1443.36	7.458	2.983	0.68	0.48	0.615	1552	862.69	1128.21	856.06
case 7	15	1578	1467.54	7.458	2.983	0.68	0.48	0.615	1578	876.38	1146.65	874.50
case 8	22	1608	1495.44	7.458	2.983	0.68	0.48	0.615	1608	893.34	1168.63	896.48
case 9	26	1628	1514.04	7.458	2.983	0.68	0.48	0.615	1628	904.45	1183.16	911.01
case 10	32	1656	1540.08	7.458	2.983	0.68	0.48	0.615	1656	919.55	1203.23	931.08

Thermal Model for Take-off at reduced thrust of 10% 27900 (lbs) / 124,1 (KN)												
	ISA DEVIATION	TET (K)	SOT (K)	φ %	m*	ε_c	ε_f	ε (overall)	Tg (K)	T cin (K)	Tb (K)	Tb (^oC)
case 1	-15	1398	1300.14	7.458	2.983	0.68	0.48	0.615	1398	776.97	1016.20	744.05
case 2	-10	1412	1313.16	7.458	2.983	0.68	0.48	0.615	1412	789.86	1029.52	757.37
case 3	-5	1435	1334.55	7.458	2.983	0.68	0.48	0.615	1435	802.77	1046.31	774.16
case 4	0	1457	1355.01	7.458	2.983	0.68	0.48	0.615	1457	815.53	1062.63	790.48
case 5	5	1481	1377.33	7.458	2.983	0.68	0.48	0.615	1481	828.72	1079.99	807.84
case 6	10	1503	1397.79	7.458	2.983	0.68	0.48	0.615	1503	841.22	1096.15	824.00
case 7	15	1526	1419.18	7.458	2.983	0.68	0.48	0.615	1526	853.76	1112.72	840.57
case 8	22	1555	1446.15	7.458	2.983	0.68	0.48	0.615	1555	870.47	1134.16	862.01
case 9	26	1574	1463.82	7.458	2.983	0.68	0.48	0.615	1574	880.68	1147.76	875.61
case 10	32	1600	1488	7.458	2.983	0.68	0.48	0.615	1600	895.4	1166.82	894.67

Thermal Model for Take-off at reduced thrust of 20% 24800 (lbs) / 110,31 (KN)													
	ISA DEVIATION	TET (K)	SOT (K)	φ %	m*	ε_c	ε_f	ε (overall)	Tg (K)	T cin (K)	Tb (K)	Tb (^{o}C)
case 1	-15	1347	1252.71	7.458	2.983	0.68	0.48	0.615	1347	755.43	983.31	711.16
case 2	-10	1370	1274.1	7.458	2.983	0.68	0.48	0.615	1370	767.91	999.84	727.69
case 3	-5	1392	1294.56	7.458	2.983	0.68	0.48	0.615	1392	780.49	1016.05	743.90
case 4	0	1415	1315.95	7.458	2.983	0.68	0.48	0.615	1415	793.62	1032.98	760.83
case 5	5	1436	1335.48	7.458	2.983	0.68	0.48	0.615	1436	805.21	1048.20	776.05
case 6	10	1458	1355.94	7.458	2.983	0.68	0.48	0.615	1458	817.75	1064.38	792.23
case 7	15	1480	1376.4	7.458	2.983	0.68	0.48	0.615	1480	830.28	1080.56	808.41
case 8	22	1511	1405.23	7.458	2.983	0.68	0.48	0.615	1511	847.15	1102.87	830.72
case 9	26	1529	1421.97	7.458	2.983	0.68	0.48	0.615	1529	857.13	1115.94	843.79
case 10	32	1554	1445.22	7.458	2.983	0.68	0.48	0.615	1554	871.72	1134.54	862.39

Thermal Model for Take-off at reduced thrust of 25% 23250 (lbs) /103,42 (KN)												
	ISA DEVIATION	TET (K)	SOT (K)	φ %	m*	ε_c	ε_f	ε (overall)	Tg (K)	T cin (K)	Tb (K)	Tb (^{o}C)
case 1	-15	1328	1235.04	7.458	2.983	0.68	0.48	0.615	1328	744.54	969.30	697.15
case 2	-10	1350	1255.5	7.458	2.983	0.68	0.48	0.615	1350	756.94	985.39	713.24
case 3	-5	1372	1275.96	7.458	2.983	0.68	0.48	0.615	1372	769.38	1001.52	729.37
case 4	0	1394	1296.42	7.458	2.983	0.68	0.48	0.615	1394	781.82	1017.64	745.49
case 5	5	1415	1315.95	7.458	2.983	0.68	0.48	0.615	1415	793.88	1033.14	760.99
case 6	10	1438	1337.34	7.458	2.983	0.68	0.48	0.615	1438	806.45	1049.73	777.58
case 7	15	1458	1355.94	7.458	2.983	0.68	0.48	0.615	1458	817.98	1064.52	792.37
case 8	22	1489	1384.77	7.458	2.983	0.68	0.48	0.615	1489	835.52	1087.25	815.10
case 9	26	1506	1400.58	7.458	2.983	0.68	0.48	0.615	1506	844.95	1099.60	827.45
case 10	32	1532	1424.76	7.458	2.983	0.68	0.48	0.615	1532	859.45	1118.53	846.38

Thermal Model for Climb, Cruise, Descent, Landing												
	ISA DEVIATION	TET (K)	SOT (K)	φ %	m*	ε_c	ε_f	ε (overall)	Tg (K)	T cin (K)	Tb(K)	Tb (^{o}C)
Climb	0	1496	1391.28	7.458	2.983	0.68	0.48	0.615	1496	835.54	1089.95	817.80
Cruise	0	1279	1189.47	7.458	2.983	0.68	0.48	0.615	1279	725.67	938.81	666.66
Descent	0	1159	1077.87	7.458	2.983	0.68	0.48	0.615	1159	667	856.52	584.37
Landing R/T	0	1413	1314.09	7.458	2.983	0.68	0.48	0.615	1413	792.75	1031.67	759.52

Appendix D

Stress model for Take-off at maximum thrust 31000 (lbs) / 137,9 (KN)											
	ISA Deviation	PCN	N (RPM)	ρ (kg/m^3)	h (m)	r cg	chord (m)	ω	ω^2	σCF (Pa)	σCF (Mpa)
case 1	-15	1.0125	15372.78	8700	0.0431374	0.3451	0.027731	1609.78	2591414.393	335624600	335.62
case 2	-10	1.021	15501.84	8700	0.0431374	0.3451	0.027731	1623.30	2635107.195	341283432.3	341.28
case 3	-5	1.0298	15635.45	8700	0.0431374	0.3451	0.027731	1637.29	2680726.933	347191829.8	347.19
case 4	0	1.0403	15794.87	8700	0.0431374	0.3451	0.027731	1653.98	2735671.838	354307967.5	354.30
case 5	5	1.0463	15885.97	8700	0.0431374	0.3451	0.027731	1663.52	2767319.181	358406743.4	358.40
case 6	10	1.0543	16007.43	8700	0.0431374	0.3451	0.027731	1676.24	2809798.755	363908445.5	363.90
case 7	15	1.0645	16162.30	8700	0.0431374	0.3451	0.027731	1692.46	2864429.477	370983892.1	370.98
case 8	22	1.0772	16355.12	8700	0.0431374	0.3451	0.027731	1712.65	2933185.249	379888731.3	379.88
case 9	26	1.0828	16440.15	8700	0.0431374	0.3451	0.027731	1721.55	2963761.806	383848825.3	383.84
case 10	32	1.0926	16588.94	8700	0.0431374	0.3451	0.027731	1737.13	3017652.279	390828399.4	390.82

Stress Model for reduced Take Off thrust of 10% 27900 (lbs) / 124,1 (KN)											
	ISA Deviation	PCN	N (RPM)	ρ (kg/m^3)	h (m)	r cg	chord (m)	ω	ω^2	σCF (Pa)	σCF (Mpa)
case 1	-15	0.9773	14838.34	8700	0.0431374	0.3451	0.027731	1553.82	2414363.185	312693979.3	312.69
case 2	-10	0.9859	14968.91	8700	0.0431374	0.3451	0.027731	1567.49	2457041.749	318221453.4	318.22
case 3	-5	0.9946	15101.01	8700	0.0431374	0.3451	0.027731	1581.32	2500597.039	323862475.7	323.86
case 4	0	1.0037	15239.17	8700	0.0431374	0.3451	0.027731	1595.79	2546564.327	329815885.9	329.81
case 5	5	1.0122	15368.23	8700	0.0431374	0.3451	0.027731	1609.31	2589878.967	335425740.8	335.42
case 6	10	1.0203	15491.21	8700	0.0431374	0.3451	0.027731	1622.18	2631495.163	340815623.3	340.81
case 7	15	1.0286	15617.23	8700	0.0431374	0.3451	0.027731	1635.38	2674483.006	346383153.4	346.38
case 8	22	1.0394	15781.21	8700	0.0431374	0.3451	0.027731	1652.55	2730940.435	353695184.2	353.69
case 9	26	1.0463	15885.97	8700	0.0431374	0.3451	0.027731	1663.52	2767319.181	358406743.4	358.40
case 10	32	1.0556	16027.17	8700	0.0431374	0.3451	0.027731	1678.31	2816732.247	364806430.2	364.80

Stress Model for reduced Take Off thrust of 20% 24800 (lbs) / 110,31 (KN)											
	ISA Deviation	PCN	N (RPM)	ρ (kg/m^3)	h (m)	r cg	chord (m)	ω	ω^2	σCF (Pa)	σCF (Mpa)
case 1	-15	0.9451	14349.45	8700	0.0431374	0.3451	0.027731	1502.62	2257887.655	292428198	292.42
case 2	-10	0.9532	14472.43	8700	0.0431374	0.3451	0.027731	1515.50	2296756.055	297462202.4	297.46
case 3	-5	0.9616	14599.97	8700	0.0431374	0.3451	0.027731	1528.86	2337414.383	302728027.4	302.72
case 4	0	0.9709	14741.17	8700	0.0431374	0.3451	0.027731	1543.64	2382845.064	308611939.5	308.61
case 5	5	0.9776	14842.90	8700	0.0431374	0.3451	0.027731	1554.29	2415845.678	312885982.9	312.88
case 6	10	0.9859	14968.91	8700	0.0431374	0.3451	0.027731	1567.49	2457041.749	318221453.4	318.22
case 7	15	0.9942	15094.93	8700	0.0431374	0.3451	0.027731	1580.69	2498586.104	323602031.5	323.60
case 8	22	1.005	15258.91	8700	0.0431374	0.3451	0.027731	1597.86	2553165.258	330670799.3	330.67
case 9	26	1.0116	15359.12	8700	0.0431374	0.3451	0.027731	1608.35	2586809.481	335028199.2	335.02
case 10	32	1.0207	15497.28	8700	0.0431374	0.3451	0.027731	1622.82	2633558.878	341082903.5	341.08

Stress Model for reduced Take Off thrust of 25% 23250 (lbs) /103,42 (KN)											
	ISA Deviation	PCN	N (RPM)	ρ (kg/m^3)	h (m)	r cg	chord (m)	ω	ω^2	σCF (Pa)	σCF (Mpa)
case 1	-15	0.927	14074.64	8700	0.0431374	0.3451	0.027731	1473.84	2172232.319	281334627.7	281.33
case 2	-10	0.9352	14199.14	8700	0.0431374	0.3451	0.027731	1486.88	2210832.29	286333866.6	286.33
case 3	-5	0.9434	14323.64	8700	0.0431374	0.3451	0.027731	1499.92	2249772.203	291377132.8	291.37
case 4	0	0.9517	14449.66	8700	0.0431374	0.3451	0.027731	1513.12	2289533.178	296526738.2	296.52
case 5	5	0.9594	14566.57	8700	0.0431374	0.3451	0.027731	1525.36	2326731.294	301344417.2	301.34
case 6	10	0.9679	14695.62	8700	0.0431374	0.3451	0.027731	1538.87	2368142.23	306707715.7	306.70
case 7	15	0.9746	14797.35	8700	0.0431374	0.341	0.027731	1549.52	2401041.225	310968598.1	310.96
case 8	22	0.9896	15025.09	8700	0.0431374	0.3451	0.027731	1573.37	2475518.498	320614452.2	320.61
case 9	26	0.9924	15067.60	8700	0.0431374	0.3451	0.027731	1577.82	2489546.91	322431328.7	322.43
case 10	32	1.0017	15208.81	8700	0.0431374	0.3451	0.027731	1592.61	2536425.731	328502795.2	328.50

Stress Model for Climb, Cruise, Descent, Landing											
	ISA DEV	PCN	N (RPM)	ρ (kg/m^3)	h (m)	r cg	chord (m)	ω	ω^2	σCF (Pa)	σCF (Mpa)
Climb	0	1.0339	15697.70	8700	0.0431374	0.3451	0.027731	1643.81	2702115.28	349961921.5	349.96
Cruise	0	0.9306	14129.29	8700	0.0431374	0.3451	0.027731	1479.57	2189136.787	283523994	283.52
Descent	0	0.7958	12082.63	8700	0.0431374	0.3451	0.027731	1265.25	1600864.862	207334508.4	207.33
Landing R/T	0	0.9693	14716.88	8700	0.0431374	0.3451	0.027731	1541.10	2374997.89	307595620.1	307.59

Appendix E

Creep Life Model Calculations							
Performance Condition	Calculated Stress	Calculated temperature	LMP	Calculated Time to Failure	t/tf (1 min)	t/tf(1.5 min)	t/tf(2 min)
take off 100% ISA -15	335.62	1043.59	26.91	610633.11	2.72941E-08	4.09411E-08	5.45881E-08
take off 100% ISA -10	341.28	1061.21	26.82	187446.24	8.89144E-08	1.33372E-07	1.77829E-07
take off 100% ISA -5	347.19	1078.51	26.76	64845.78	2.5702E-07	3.8553E-07	5.1404E-07
take off 100% ISA 0	354.30	1093.96	26.73	27166.24	6.13507E-07	9.2026E-07	1.22701E-06
take off 100% ISA 5	358.40	1111.89	26.7	10305.81	1.61721E-06	2.42581E-06	3.23442E-06
take off 100% ISA 10	363.90	1128.21	26.64	4096.64	4.06837E-06	6.10255E-06	8.13673E-06
take off 100% ISA 15	370.98	1146.65	26.57	1485.32	1.12209E-05	1.68313E-05	2.24418E-05
take off 100% ISA 22	379.88	1168.63	26.5	474.28	3.51407E-05	5.27111E-05	7.02814E-05
take off 100% ISA 26	383.84	1183.16	26.47	235.57	7.07503E-05	0.000106125	0.0001415
take off 100% ISA 32	390.82	1203.23	26.41	88.94	0.00018739	0.000281084	0.000374779

Creep Life Model Calculations

Performance Condition	Calculated Stress	Calculated Temperature	LMP	Calculated time to failure	t/tf (1 min)	t/tf (1.5 min)	t/tf (2 min)
take off 90% ISA -15	312.69	1016.20	27.11	4761227.45	3.5005E-09	5.25075E-09	7.001E-09
take off 90% ISA -10	318.22	1029.52	27.06	1923345.53	8.66546E-09	1.29982E-08	1.73309E-08
take off 90% ISA -5	323.86	1046.31	27.01	652115.28	2.55579E-08	3.83368E-08	5.11157E-08
take off 90% ISA 0	329.81	1062.63	26.95	229835.98	7.25155E-08	1.08773E-07	1.45031E-07
take off 90% ISA 5	335.42	1079.99	26.89	79128.51	2.10628E-07	3.15942E-07	4.21256E-07
take off 90% ISA 10	340.81	1096.15	26.85	31245.49	5.3341E-07	8.00115E-07	1.06682E-06
take off 90% ISA 15	346.38	1112.72	26.8	12164.75	1.37008E-06	2.05512E-06	2.74016E-06
take off 90% ISA 22	353.69	1134.16	26.73	3698.21	4.50668E-06	6.76002E-06	9.01336E-06
take off 90% ISA 26	358.40	1147.76	26.69	1794.57	9.28727E-06	1.39309E-05	1.85745E-05
take off 90% ISA 32	364.80	1166.82	26.64	677.88	2.45864E-05	3.68797E-05	4.91729E-05

Creep Life Model Calculations

Performance Condition	Calculated Stress	Calculated Temperature	LMP	Calculated time to failure	t/tf(1 min)	t/tf(1.5 min)	t/tf(2 min)
take off 80% ISA -15	292.42	983.31	27.3	57972694.56	2.87492E-10	4.31238E-10	5.74983E-10
take off 80% ISA -10	297.46	999.84	27.24	17544793.24	9.4995E-10	1.42492E-09	1.8999E-09
take off 80% ISA -5	302.72	1016.05	27.19	5758672.07	2.89419E-09	4.34128E-09	5.78837E-09
take off 80% ISA 0	308.61	1032.98	27.15	1918494.54	8.68737E-09	1.30311E-08	1.73747E-08
take off 80% ISA 5	312.88	1048.20	27.11	729955.65	2.28324E-08	3.42487E-08	4.56649E-08
take off 80% ISA 10	318.22	1064.38	27.06	264889.92	6.29192E-08	9.43788E-08	1.25838E-07
take off 80% ISA 15	323.60	1080.56	27.02	101255.37	1.646E-07	2.469E-07	3.29201E-07
take off 80% ISA 22	330.67	1102.87	26.95	27292.57	6.10667E-07	9.16E-07	1.22133E-06
take off 80% ISA 26	335.02	1115.94	26.91	13002.19	1.28183E-06	1.92275E-06	2.56367E-06
take off 80% ISA 32	341.08	1134.548	26.86	4727.30	3.52562E-06	5.28843E-06	7.05123E-06

Creep Life Model Calculations							
Performance condition	Calculated stress	Calculated temperature	LMP	Calculated time to failure	t/tf (1 min)	t/tf (1.5 min)	t/tf (2 min)
take off 75% ISA -15	281.33	969.30	27.37	172520539	9.66069E-11	1.4491E-10	1.93214E-10
take off 75% ISA -10	286.33	985.39	27.35	56917926.34	2.92819E-10	4.39229E-10	5.85639E-10
take off 75% ISA -5	291.37	1001.52	27.3	18134898.59	9.19039E-10	1.37856E-09	1.83808E-09
take off 75% ISA 0	296.52	1017.64	27.24	5857256.69	2.84547E-09	4.26821E-09	5.69095E-09
take off 75% ISA 5	301.34	1033.14	27.2	2124662.63	7.84438E-09	1.17666E-08	1.56888E-08
take off 75% ISA 10	306.70	1049.73	27.16	746729.48	2.23196E-08	3.34793E-08	4.46391E-08
take off 75% ISA 15	310.96	1064.52	27.11	292858.38	5.69103E-08	8.53655E-08	1.13821E-07
take off 75% ISA 22	320.61	1087.25	27.04	74128.96	2.24833E-07	3.3725E-07	4.49667E-07
take off 75% ISA 26	322.43	1099.60	27.01	36599.01	4.55386E-07	6.83079E-07	9.10771E-07
take off 75% ISA 32	328.50	1118.53	26.95	12419.80	1.34194E-06	2.01291E-06	2.68389E-06

Creep Life Model Calculations					
Flight segment	Calculated stress	Calculated temperature	LMP	Calculated time to failure	t/tf
Climb	349.96	1089.95	26.78	37136.1888	8.97597E-07
Cruise	283.52	938.81	27.37	1424509496	1.75499E-09
Descent	207.33	856.52	28.21	8.61914E+12	2.32042E-14
Landing Reverse Thrust	307.5956201	1031.676769	27.15	2071963.21	1.34065E-09